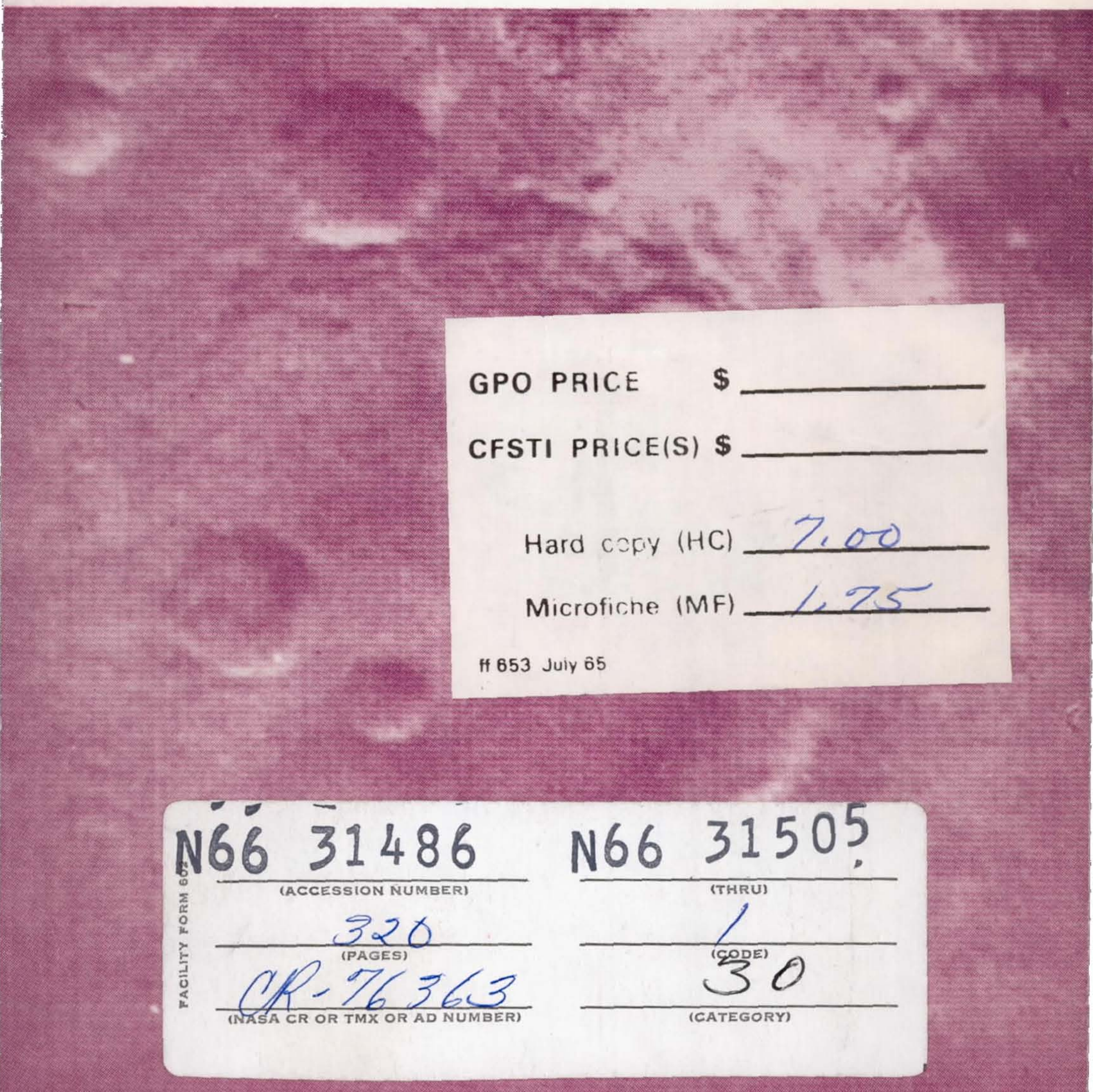
CONFERENCE ON THE EXPLORATION OF MARS AND VENUS



VIRGINIA POLYTECHNIC INSTITUTE
Blacksburg, Virginia
August, 1965

FRONT COVER

MARINER IV, PICTURE 11, JULY 24, 1965. THE CENTER OF THE PICTURE IS IN ATLANTIS - BETWEEN MARE SIRENUM AND MARE CIMMERIUM, 31° S. LAT., 197° E. LONG. THE AREA COVERED IS 25,500 MILES. PHOTO USED BY PERMISSION OF JET PROPULSION LABORATORY, PASADENA, CALIFORNIA.

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VENUS PHOTOGRAPHED IN RED AND IN YELLOW LIGHT AT INFERIOR CONJUNCTION, JUNE 19, 1964. UPPER PICTURE IS A COMPOSITE OF 5 PHOTOGRAPHS TAKEN IN RED LIGHT; THE LOWER PICTURE IS A COMPOSITE OF 2 PHOTOGRAPHS TAKEN IN YELLOW LIGHT. PHOTOS COURTESY OF LOWELL OBSERVATORY.

COVER DESIGNED BY DAVID A. WILLMENT
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VIRGINIA POLYTECHNIC INSTITUTE

ENGINEERING EXTENSION SERIES CIRCULAR No. 5

PROCEEDINGS OF THE CONFERENCE ON

The Exploration of Mars and Venus

AUGUST 23-27, 1965

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INTRODUCTION

Although the lunar landing and exploration effort will continue for some time to absorb a large part of the total space exploration effort, attention is now being focussed on the possibilities of exploration of the near earth planets, Mars and Venus.

While astronomical observations of these planets has been going on for considerable time, recent years have seen an increase in this activity. One Venus "Fly-By" has been accomplished and the success of Mariner IV gives evidence that this means of observation is practical and undoubtedly will continue. The problems of manned and unmanned exploration of these planets have occupied the attention of a number of scientists and engineers engaged in the space effort.

The papers presented at the conference discussed the characteristics of the planets and their atmospheres and the problems inherent in their exploration. The topics included information, speculation and future planning based on the best opinions of experts in the field.

It was the aim of the conference to furnish participants from educational institutions, private industry, research laboratories and governmental agencies knowledge of this next step in space exploration. For the educators, perhaps new information for instruction and new avenues of research were the greatest dividends to be expected. In giving the other participants an opportunity to hear the leaders in the field and discuss their own problems, it is hoped that the Conference made a significant contribution to the space exploration effort.

ACKNOWLEDGEMENT

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VPI particularly wishes to acknowledge the assistance of the Langley Research Center of NASA in helping in planning the conference and in providing speakers and session chairmen. VPI would also like to thank the following organizations that provided speakers for the conference.

Colorado State University
Jet Propulsion Laboratory
Johns Hopkins University
NASA - Ames Research Center
NASA - George C. Marshall Space Flight Center
NASA - Office of Space Science and Applications
The Rand Corporation
The University Newcastle Upon Tyne
Yale University Observatory

The co-operation of Poly-Scientific Division, Litton Precision Products, Inc. in providing a tour of their facilities for the manufacture of slip rings and torque motors, in providing transportation to the banquet and for sponsoring the reception before the banquet is greatfully acknowledged.

The Conference Committee also wishes to express its appreciation to the faculty and administration of VPI and to the speakers and session chairmen all of whom contributed to the success of the conference.

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THE ORBITS AND THE GRAVITATIONAL FIELDS OF MARS AND VENUS

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INTRODUCTION

The object of this paper is to present some general information on the orbits of Mars and Venus, on the orbits of the Martian satellites, and on the gravitational fields of these planets obtained by application of methods of dynamical astronomy.

The two planets present great contrasts: owing to the presence of well observable surface markings on Mars, the rotation period of this planet is known with a high degree of accuracy. The careful study of Ashbrock (1953) yields for the sidereal period of rotation, expressed in ephemeris days,

$$24^h\ 37^m\ 22^s.6689 \pm 0^s.0026\ (\text{s.d.}).$$

The orientation of the axis of rotation of Mars is well known, both from its surface markings and from observations of the satellites.

Venus lacks distinct surface markings. As a consequence optical observations have not yielded anything definite on either the period or the orientation of the axis of rotation. Recent radar Doppler observations indicate a period of rotation of 250 ± 50 days in retrograde direction. Furthermore, Venus lacks satellites, and therefore all the advantages to an astronomer of a planet attended by satellites.

HISTORICAL NOTES

Historically the orbit of Mars is the most famous of all the classical planetary orbits. It is from the study of the observations of Mars by Tycho Brahe that Kepler succeeded in deducing his first two laws of planetary motion:

1. The orbit of each planet is an ellipse with the sun in one focus.
2. The straight line joining the sun and a planet sweeps out equal areas in equal intervals of time.

It is well known that Kepler first attempted to satisfy Tycho's observations by making use of the time honored scheme of epicycles, such as Copernicus had continued to use after introducing the heliocentric theory of planetary motion. Only after many attempts had failed did Kepler decide to make a new start in which two provisional assumptions were made: (1) that the earth moves in a known circular orbit around the sun such that the position in the orbit can be calculated for any date; (2) that Mars moves in a periodic orbit the period of which was well enough known. From the rich observational material left to him by Tycho Brahe, Kepler could choose many pairs of observations taken one period of revolution of Mars apart. From the quadrangle formed by the sun, the two positions of the earth and the two identical positions of Mars in its orbit, the line sun-Mars for that point in the orbit became known both in length (in units of the radius of the earth's orbit) and in direction. By repeating the same process with numerous pairs of observations the true character of the Martian orbit became apparent to Kepler. Even then it took him some time to recognize that the figure of the orbit was an ellipse with the sun in one focus, and that the rate of motion was in accordance with the law of equal areas in equal time intervals.

It may be said that Kepler was lucky in choosing the orbit of Mars for his exhaustive study. The eccentricity of the orbit (0.093) is large enough to make apparent from Kepler's construction the difference between an eccentric circle and an ellipse. A fuller examination of the circumstances is necessary.

It had been known since Copernicus that the planets, including the earth, revolved around the sun in paths that were not very different from circles. The deviations from circular motion could be represented roughly by systems of eccentrics and epicycles. These diviations were notably different in amount for different planets, being very small in the case of Venus, relatively large in the case of Mars, still larger in that of Mercury. The Prussian Tables calculated by Reinhold on a Copernican basis, published in 1551, were found to represent the actual motions so imperfectly that errors of 4° or 5° were noted by Tycho and Kepler. The solution of the problem was clearly more likely to be found by the study of a planet in which the deviations from circular motion were as great as possible. In the case of Mercury satisfactory observations were scarce, whereas in the case of Mars an abundant series of observations had been recorded by Tycho. Hence it was true insight on Tycho's part to assign to his ablest assistant this particular planet and on Kepler's part to continue his research efforts with exceptional patience.

An ellipse is only a first approximation to a planetary orbit. If an attempt were made to represent modern planetary observations with the aid of elliptic orbits, the inadequacy of the representation would soon be apparent. The good fortune of Kepler was that Tycho's observations were just accurate enough to reveal the elliptic character of the orbit. If they had been too accurate, the perturbations would have shown up, and Kepler's effort might well have bogged down in endless experimentations that could lead nowhere.

PLANETARY THEORIES

Laplace, in the preface to the Mécanique Céleste, wrote:

"Astronomy, considered in the most general way, is a great problem of mechanics, the arbitrary data of which are the elements of the celestial movements; its solution depends both on the accuracy of observations and on the perfection of analysis."

This, of course, represents the position of celestial mechanics after Newton had laid its foundations. All work on planetary and satellite motion before Newton was empirical in nature.

It is perhaps remarkable that the great mathematical astronomers of the 18th century who first employed the powerful tools of analysis to the solution of the problems of the motion of bodies in the solar system were attracted more to the problem of the moon's motion than to that of planetary motion. The great step forward was made by Laplace (1749-1827), who in 1786 discovered the nature of the long-period inequalities in the motions of Jupiter and Saturn, a phenomenon which is the clue to an important feature of planetary theory. In other areas of planetary theory Laplace's contributions are equally outstanding.

Following Laplace, the names of Hansen (1795-1874), Leverrier (1811-1877) and Simon Newcomb (1835-1911) are the great names in planetary theory. Leverrier provided the theories of the motions of all the principal planets, Mercury to Neptune, by the method of the variation of elements. For Uranus and Neptune his work was extended by Gaillot. Newcomb proceeded independently, using a method of perturbation in polar coordiantes for the four inner planets and for Uranus and Neptune. The theories of Jupiter and Saturn were treated by Hill, who used Hansen's method.

The only published new work on the theories of the principal planets is the new theory of Mars by Clemence (1949, 1962). The comparison of the first-order theories by Clemence and Newcomb shows excellent agreement in general. The principal improvement of Clemence's new theory over Newcomb's work lies in the fact that Clemence introduced second-order terms and even some third-order terms that Newcomb has omitted. The orbit of Mars is so powerfully perturbed by Jupiter and the earth that in order to obtain a theory of the planet to satisfy the requirements of modern observational accuracy, the second-order perturbations should be explored more fully than Newcomb did. Newcomb was, in fact, aware of this, as is apparent from his introduction to the Tables of Mars, where he refers to

"---a great number of minute terms depending on the product of the masses of Jupiter and Saturn, which, while individually too small to be important, might in the aggregate occasionally attain an appreciable magnitude."

The comparison with observations is the ultimate test of a planetary theory; in the case of the new theory of Mars the comparison was supplemented by the comparison with a numerical integration computed by Herget with the Naval Ordnance Research Calculator. Although this comparison extends over only 35 years, it provides a satisfactory proof of the accuracy of the general theory. The differences in latitude never exceed $0''.008$; in longitude the largest difference is $0''.04$. This represents a new standard of accuracy for a general planetary theory that surpasses previous efforts by a considerable margin.

An exhaustive discussion of the available observations since about 1750 is being carried out at the U.S. Naval Observatory by Duncombe. However, a provisional ephemeris based on Clemence's new theory, with the constants obtained from a comparison with 87 observations made in the years 1802-1839 and 1931-1950, has been published in U.S. Naval Observatory Circulars for the two hundred years 1800-2000. According to Dr. Duncombe the uncertainty in the printed rectangular coordinates is not expected to exceed a few units in the seventh decimal, or about two percent of the diameter of Mars.

No new theories have so far been constructed for the other inner planets, but Dr. Clemence has undertaken the development of a new theory of the earth's orbital motion by Hansen's method. This work is far advanced. Of all the orbits of the principal planets, that of the earth is of the greatest importance because the interpretation of all observations made from observatories located on the surface of the earth requires an accurate knowledge of the earth's orbit. In this connection it is of interest also to note a remark by Clemence that further refinement of the theory of Mars can be accomplished only if the theory of the earth's orbital motion is improved.

Newcomb's Tables of Venus were compared with observations by Duncombe (1958). The merit of this work is that it adds nearly sixty years of observations to the data that were available to Newcomb. The secular changes in the elements are therefore determined with considerably greater accuracy than in Newcomb's discussion. Especially noteworthy is that a discordance between the observed and theoretical value of the motion of the node of the orbit of Venus obtained by Newcomb is not confirmed by Duncombe. The probable explanation is that the discordance was caused by systematic errors in the older observations.

TYPICAL PLANETARY PERTURBATIONS

Tables I and II list the principal first-order periodic perturbations in the longitudes of Venus and Mars. They consit of sine terms the arguments of which are of the form

$$jl' + kl + \text{const.},$$

l, l' being the mean anomalies of the perturbed planet and the perturbing planet, respectively; in the second-order perturbations terms of the form

$$\Sigma\text{coeff.} \times t \sin (jl' + kl + \text{const.})$$

and

$$\Sigma\text{coeff.} \times \sin (j_1 l' + j_2 l'' + kl + \text{const.})$$

appear as well as contributions factored by t^2 as additions to the secular terms.

The perturbing planet is designated at the head of each column. The letters V, E, M, J, S in the arguments are used as abbreviations of the mean anomalies of the planets Venus, earth, Mars, Jupiter, and Saturn. A constant part, different for each argument, is to be added.

Some of these periodic perturbations call for comments:

The 13E - 8V term in the longitude of Venus is a long-period term with period 239 years arising from the near commensurability 13/8 between the mean motions of Venus and the earth.

Similarly in the longitude of Mars the 3M - V term is a long period term with period 33 years arising from the near commensurability 3/1 between the mean motions of Venus and Mars. This particular term may be used for obtaining a determination of the mass of Venus.

The term with argument 2M - E has been magnified by the near-commensurability 2/1 between the mean motion of the earth and Mars. Its period is 16 years. A closer approximation is 15/8, hence the significance of the term 15M - 8E, which has the seventh power of the eccentricity as a factor. Its period is 40 years.

In the conventional form of planetary theory powers of the time are permitted to appear in the coefficients of periodic terms. Such planetary theories are valid for a limited span of time, perhaps ten centuries, depending on the extent to which terms of higher order have been included. In any case, the quality of the representation of the observations is bound to diminish as the interval of time from the epoch of the theory is increased.

In principle it is possible to develop formal solutions of the equations of planetary motion free from this handicap. The theory of the secular variations of planetary elements serves to indicate the type of solution to be expected. Although these results are of considerable usefulness for certain lines of investigation, they are not comparable with complete planetary theories.

THE SATELLITES OF MARS

Mars has two known satellites, Phobos and Deimos. Both were discovered by Asaph Hall Sr. at the U.S. Naval Observatory in August 1877 during the particularly favorable opposition of that year.

The periods of revolution are:

Phobos	$7^h.65385$	= 0.31084 P
Deimos	$30^h.29858$	= 1.23050 P

P being the period of axial rotation of Mars.

Thus Phobos moves eastward among the stars 3.217 times as fast as the apparent westward motion of the stars (for a Martian observer) on account of the rotation of the planet. The satellite thus moves from west to east in the Martian sky and crosses each meridian 2.217 times during a Martian sidereal day.

Deimos' period of revolution is longer than the period of rotation of the planets. Hence this satellite moves from east to west in the Martian sky. However, since its sidereal motion is 0.813 times the apparent westward motion of the stars, its westward motion relative to a Martian meridian is only 0.187 times the westward motion of the stars. Hence it takes 1/0.187 = 5.34 Martian days to come back to the meridian (compared with $24^h51^m$for the earth moon).

The scale of the orbits of these satellites, especially of the outer one, yields a good value of the mass of the planet (in terms of the sun's mass). Hall's value obtained from the observations of 1877 was adopted by Newcomb in his work on the theories of the inner planets and has been in general use since. This mass ratio is

$$\frac{M_{Mars}}{M_{sun}} = \frac{1}{3,093,500}$$

Newcomb comments on this mass determination:

"When nearest the earth, the major axis of the orbit of the outer satellite subtends an angle of 70". I can not think that the systematic error to be feared in the best measures, such as those made by Prof. Hall, can be as great as half a second. It therefore appears to me that the mean error in adopting Prof. Hall's value of the mass does not exceed its fiftieth part. This is a degree of precision much higher than that of any determination through the action of Mars on another planet".

Recent determinations have confirmed this value of the reciprocal of the mass of Mars, viz.

Rabe (1950)	3,110,000 ± 7,700
Mariner IV (1965)	3,098,600 ± 3,000

Rabe's determination was a by-product of his discussion of observations of Eros whose main object was a determination of the mass ratio (Earth + Moon) /Sun, which yields the dynamical determination of the solar parallax (or of the astronomical unit expressed in kilometers). It is well known that Rabe's value of the astronomical unit expressed in kilometers disagrees with the value obtained from radar echo observations of Venus and confirmed by similar observations of Mercury. Marsden (1965) found that he could bring Rabe's and the radar echo determination into agreement if the reciprocal of the mass of Mars were decreased to 3,020,000. This value would differ from Hall's value by one part in 43, not far from Newcomb's estimate of the maximum error in Hall's value. The provisional result obtained by the Mariner IV prove flying by Mars renders Marsden's value doubtful.

Table I. Principal periodic perturbations in longitude, Venus

Earth	coeff.	Mars	coeff.
+ 1 E - 1V	4".89	+3M - 1V	1".21
+ 2 - 2	11.26		
+ 3 - 2	3.45		
+ 4 - 4	1.03	Jupiter	coeff.

+ 5	- 4	1.58	+1J - 1V	2".97
+ 5	- 3	1.44	+1 0	1.56
+13	- 8	2.79		

Table II. Principal periodic perturbations in longitude, Mars

Venus		coeff.	Jupiter	coeff.
- 1 V	+ 3M	6".37	+1J - 2M	3".14
			+1 - 1	25.38
Earth			+1 0	3.73
- 1 E	+ 1M	8".56	+2 - 3	2.11
- 1	+ 2	13.97	+2 - 2	16.04
- 2	+ 3	7.36	+2 - 1	21.87
- 2	+ 4	4.91	+3 - 3	1.31
- 3	+ 5	2.64	+3 - 2	2.61
- 8	+15	1.55	+3 - 1	3.17
			Saturn	coeff.
			+1 S - 1M	1".35
			+2 - 1	1.77

THE OBLATENESS OF MARS

The principal perturbations in the motions of the satellites are those arising from the oblateness of the planet. Woolard (1944) in a discussion of the data ascribes to this cause the annual motions of the ascending node

Phobos	-158°.5	± 0°.5
Deimos	- 6°.2795	± 0°.0007

With R = 4".680 at distance 1 astr. unit he derives

$$J_2 = 0.001947 \quad .$$

For a rotating body whose surface is an equipotential surface the first-order relation

$$f = \frac{1}{2}(3 J_2 + \phi)$$

applies, in which

$$\phi = \frac{\omega^2 R^3}{f\, m_p} = \frac{\omega^2 R}{g}$$

the ratio between centrifugal acceleration and acceleration of gravity at the equator. Moreover, for such a rotating body,

$$0.50 < \frac{f}{\phi} < 1.25$$

The lower limit pertains to a body with all its mass concentrated at its center, the upper limit to a homogeneous body. The observational data give for Mars

$$\phi = 0.004548$$

and hence

$$f = 0.005209$$

$$f/\phi = 1.145 \quad .$$

Values for other planets are

	f/ϕ
Earth	0.969
Jupiter	0.774
Saturn	0.688
Neptune	0.88

The dynamically obtained value of f/ϕ for Mars is reasonable. With its smaller mass, Mars is likely to be more nearly homogeneous than the earth is. On the other hand, optical determinations of the oblateness of Mars are sytematically greater than the dynamical determination, by a factor of almost two. With $f = 0.0100$ there would result

$$f/\phi = 2.20 \quad ,$$

much larger than for a homogeneous body. Two possible explanations may be considered: (a) the surface of Mars is far from an equipotential surface, (b) the optical determinations are affected by systematic errors. The latter of these is perhaps the more likely, in view of the difficulty of the observations, affected as they may be by phase effects and polar caps. If this should be so, it is hard to explain why optical determinations by different methods should be so accordant, yet different from the dynamical value.

THE MASS OF VENUS

Newcomb's value for the reciprocal of the mass of Venus in terms of the sun's mass is

$$m^{-1} = 408{,}000 \quad .$$

Of the determinations made since Newcomb's time the three most important ones are with their porbable errors: (1) that by Morgan and Scott (1939) from periodic perturbations by Venus on the earth

$$m^{-1} = 407{,}000 \pm 500 \quad ,$$

(2) that by Rabe (1950) from the perturbations by Venus on the motion of Eros

$$m^{-1} = 408{,}645 \pm 208 \quad ,$$

(3) that from Mariner II, by Anderson, Null and Thornton (1964)

$$m^{-1} = 408{,}539.5 \pm 12 \quad .$$

The accordance between Rabe's value and that obtained from Mariner II is pleasing. The rounded number,

$$m^{-1} = 408{,}540 \pm 18 \text{ (s.d.)}$$

may serve as the best value currently available.

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BALLOON - TELESCOPE OBSERVATION OF THE PLANETS

By

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INTRODUCTION

In February of this year a determination of water vapor in the upper atmosphere of Venus was made. The amount of water found above the planet's clouds is of the same order of magnitudes as in the earth's stratosphere. This measurement settles the question of the presence of water on Venus.

The quantity measured is in agreement with the upper limit of the 1959 measurements, from a manned balloon flight of Moore and Ross. The accuracy, however, is much greater. The relative error is only 5%.

The data were obtained by an automatic telescope-spectrometer unit, carried by balloon to an altitude of 27 km. There was no observer on board. The telescope, of 30 cm aperture, was pointed at Venus by the tracking system that was described by Strong and Bottema in last year's Colloquium Reports [1]. An overall picture of the unit, as it ascended from Holloman Air Force Base, New Mexico, on February 21, 1964, is given in Figure 1. Figure 2 shows the equipment inside the dome.

This paper discusses some aspects of the data recording system, the data reduction and the interpretation of the results in the light of our present knowledge of the atmosphere of Venus.

DATA RECORDING

The radiation was measured in the band at 1.13 microns, with a grating spectrometer of 2 Å resolving power. The spectrum was imaged upon a thin stainless steel strip perforated by 21 exit slits, each of 1.4 Å spectral width, placed at positions corresponding with minima in the water-vapor spectrum, as shown in Figure 3. The positions of the slits are indicated by numbers 1 through 15, and 18 through 23. In the table, the positions are in the same order, and designated by letters.

While sampling the data, the array of slits was driven back and forth about the spectrum-matching position over a range equivalent to a spectral width of 16.7 Å. Each scan had a time duration of 10 seconds. Water-vapor absorption, in the optical path, was manifest as a minimum at the spectrum-matching position, the depth of the minimum being a measure for the amount of water-vapor penetrated by the radiation. An advantage of scanning the slits is that slight changes in wavelength calibration, brought about by the environment at altitude, are eliminated. In the off-match position some slits may still be aligned with regions of absorption, which leads to a reduction of the modulation. This effect, however, is very small for the displacements used, and is easily accounted for by calibration.

The radiation entering the spectrometer was chopped at a frequency of 30 cps. The detector used was a photomultiplier with an S-1 photocathode, selected for sensitivity in the near infrared. The AC component of its response was amplified and recorded on photographic paper by mirror galvanometers. A typical section of the flight record is shown in Figure 4.

Three galvanometers, with the same zero line but differing sensitivities, were used to assure a wide dynamic range. The strip on which galvanometer deflections were recorded also carried records of the error signals of the Venus tracker. These records were useful to identify spurious changes in signal level, caused by small irregularities in tracking. From 120 scans, selected for high-tracking stability, the modulation due to water vapor in the path was derived to be (10.5 ± 0.5) %. No systematic variation in the modulation depth appeared during the two hours of observation. The observation period extended from 2 to 4 o'clock in the afternoon, and Venus culminated around 3 o'clock.

In addition to the water-vapor minimum, another dip in galvanometer response occurs at the point in the scan where slit "p" (in Table I) came into position to pass the mercury emission line at 11287 Å. This was produced by radiation from a low pressure mercury arc lamp. The lamp was positioned near the entrance slit, so that its emission was scattered into the spectrometer from the back of the chopper blade. The mercury line, when it fell on slit "p", appeared as a dip in the recorded signal since its radiation was received in the negative half-cycle of the chopper.

The recorded data thus had two dips during each scan: one resulting from absorption by water vapor, and one resulting from the off-cycle mercury emission. The position of the water-vapor dip with respect to the mercury dip gives us a measure of the Doppler shift in the radiation received from Venus. Twenty of the cleanest scans were selected and carefully measured. The Doppler shift thus found was (0.49 ± 0.05) A°, in excellent agreement with the value calculated from the relative orbital motions of Venus and the earth, which at the time of observation was 0.495 A°.

DATA REDUCTION

The instrument was calibrated at sea level pressure before flight, in a room with measured humidity. The calibration curve is shown in Figure 5.

This calibration is in agreement with calculations of Dr. William S. Benedict, of this laboratory, for the resolving power of our instrument. Dr. Benedict's calculations, made for our less successful 1959 flight, were for the latter two-thirds of the 21 slits in our array, but the seven slits which have been added correspond to absorption lines of similar strengths. The 10.5% modulation, found in flight, corresponds to 9.8×10^{-3} g/cm^2 of precipitable water at one atmosphere, at room temperature.

The amount of water vapor detected must still be corrected for residual water vapor in the earth's atmosphere above the level of the balloon. At the end of the Venus measurement, doors closed over the telescope. The doors were provided with an opening covered with thin white cloth, to receive sunlight and scatter it into the telescope. The modulation thus measured was less than 1.0%. A typical scan is shown in Figure 6. The solar elevation at this time was about 20°. The elevation of Venus, at culmination, was 62°. It seems therefore that at the time of the Venus observation the terrestrial contribution in the modulation was about 0.5%.

Both assessments of terrestrial water vapor, by Doppler shift and by sunlight measurement, are in fair agreement with the currently accepted value of about 7×10^{-4} g/cm^2 of water above the altitude of the balloon. The extension of our laboratory water-vapor calibration to the lower pressures at altitude is based again upon the calculations made by Dr. Benedict.

Since the Doppler shift between Venusian and terrestrial water absorptions is sufficiently greater than the equivalent width of either, we may directly subtract out the modulation measured with sunlight from the modulation measured with Venus, even though we are not operating in the linear absorption region. The Venus water dip is thus corrected from 10.5% to 10.0%.

INTERPRETATION

To interpret the absorption in terms of water-vapor quantities on Venus, it is necessary to assume a range of pressures from the cloud deck outward. In the following, we have supposed that the water vapor above the visible clouds is distributed gravitationally, with a uniform mixing ratio, For the base pressure we have used the limits reported in the survey article by Sagan [9] : 90 mb to 600 mb at the cloud level. To treat the gravitational pressure distribution we assume a Lorentz line shape, and perform two integrations. One integration is over the altitude above the Venusian cloud deck, from zero to infinity. The other integration is over wavelength, from the center of the absorption out to the edges of the slit, which may be taken as ± infinity. These integrations yield a correction factor, which must lie between 1 and $\sqrt{2}$, by which the measured 10.0% modulation must be multiplied before reading the calculated calibration curve for a uniform atmosphere at the base pressure. For the 600 mb base pressure the absorption measured is nearly in the linear region, and the result may be taken as 1.1×10^{-2} g/cm^2. The 90 mb case is nearly in the square-root region, and the result is 4.7×10^{-2} g/cm^2.

At the time of our observation the phase angle of Venus was 65°. The average slant path through the Venusian atmosphere, assuming cosine scattering from the cloud deck and integrating over the visible surface, was 3.82 times the vertical path through the atmosphere.

When the values found are divided by 3.82 for the slant path correction, the results are 12.3×10^{-3} g/cm^2 for the 90 mb case, and 2.9×10^{-3} g/cm^2 for the 600 mb case.

A choice between these values, or in this range, must await more knowledge about the actual pressures. It is interesting, however, to note that values reported for comparable levels in the earth's upper atmosphere [10] represent the geometric mean of the extremes we have calculated for the planet Venus.

COMPOSITION OF THE VENUS CLOUDS

Solar radiation reflected by the Venus clouds in the infrared spectrum region 1.7 to 3.4 microns [2] was measured on a subsequent flight (on October 28, 1964). This Venus spectrum, Curve 1; and the reflection spectrum of a laboratory ice cloud, Curve 3, are shown in Figure 7. From the similarity we conclude that Venus clouds are composed of ice crystals.

The remaining difference between Curves 1 and 3 of Figure 7 can be accounted for by correction for the residual vapor absorption in the upper atmosphere of Venus. Corrections for the absorption yield the reflection spectrum of the Venus cloud deck itself--Curve 2.

Two vapors are important to the above corrections: one of these, carbon dioxide, has been measured with ground-based instruments, [3] and the corrections are easily made at 2.0 and 2.8 microns. The other absorbing vapor is water. To correct for it we invoke the water vapor quantity measured on the previous flight. [1] From this we estimate absorptions at 1.9 microns and near 2.7 microns, by means of the laboratory studies of Howard, Burch, and Williams. [4] With these corrections no discrepancies remain.

The quality of the agreement at 2.6 microns may be taken as a direct confirmation of the prior determination [1] of the quantity of water vapor. Similar agreement exists at 1.9 microns. The laboratory cloud spectral feature near 2.6 microns varies slightly with cloud conditions. One may conclude, within limits set by the uncertainties of the agreement, that the effective cloud reflection levels for radiations of 1.1, 1.9, and 2.6 microns all lie at essentially the same altitude. The absorption correction indicates a probably pressure near 100 mb, low in the range of pressures (90 to 600 mb) that we used to interpret our observations. [1]

The identification of the Venus clouds as ice particles suggests an explanation of a phenomenon that must be accounted for by any satisfactory model of the Venus atmosphere: the temperature at the Venus cloud surface lies near -40°C. This observed temperature is the temperature at which supercooled water vapor spontaneously freezes in the absence of nuclei of condensation. [5] The cloud temperature does not vary significantly from the bright side to the dark side. [6] Considering the measured slow rotation of the planet, a substantial mechanism is necessary to carry approximately half the absorbed solar flux across the terminator to be re-radiated from the visually dark hemisphere in order to sustain -40°C temperatures there. The latent heats of condensation and freezing of water provide this mechanism. Water vapor from the sunlit hemisphere, as it is carried convectively to the dark hemisphere, will cool, condense and freeze, with the release of over 600 calories/gram, to prevent the cloud temperatures on the dark hemisphere of the planet from falling below -40°C. The water is again evaporated after it is returned to the sunlit hemisphere. A reasonable burden of atmospheric water and reasonable wind speeds, such as are frequently observed on the earth, are adequate for this mechanism.

The positive identification of the cloud particles as water and ice requires that all of our knowledge of terrestrial clouds be applied to the discussion of Venus, particularly in relation to the anomalously high microwave brightness. Terrestrial observations show that clouds emit non-thermal microwave radiation. [7] Furthermore, Tolbert and Straiton have pointed out [8] that this source of microwaves should have a spectral distribution which copies that of thermal radiation, and is a likely candidate for explaining the high microwave brightness. We feel that this spherics mechanism has not yet been adequately considered. The actual surface temperatures on Venus may well be tolerably low.

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TABLE I

Positions of Exit Slits

a. 11149 Å	h. 11218 Å	o. 11276 Å
b. 11164	i. 11222	p. 11295
c. 11171	j. 11225	q. 11322
d. 11181	k. 11235	r. 11332
e. 11187	l. 11252	s. 11337
f. 11201	m. 11260	t. 11345
g. 11211	n. 11271	u. 11358

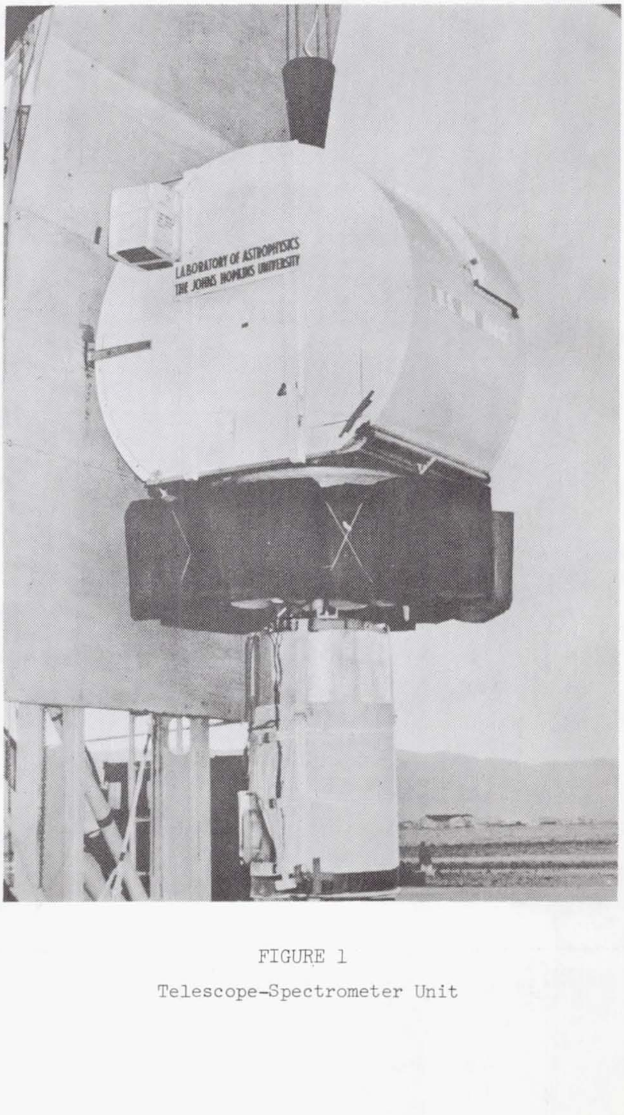

FIGURE 1

Telescope-Spectrometer Unit

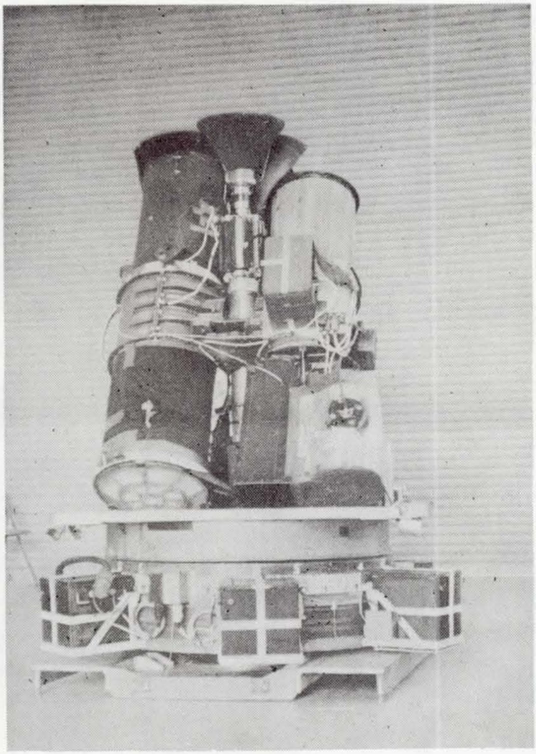

FIGURE 2

Telescope-Spectrometer Unit

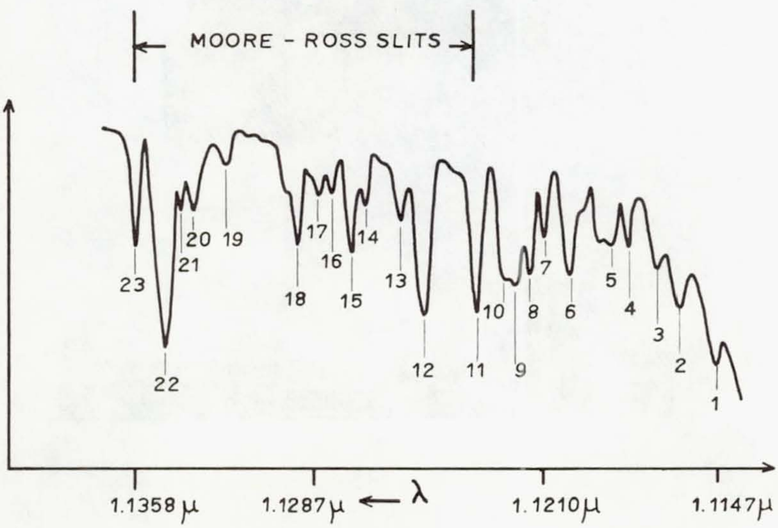

FIGURE 3

Single Slit Scan Across H_2O Band

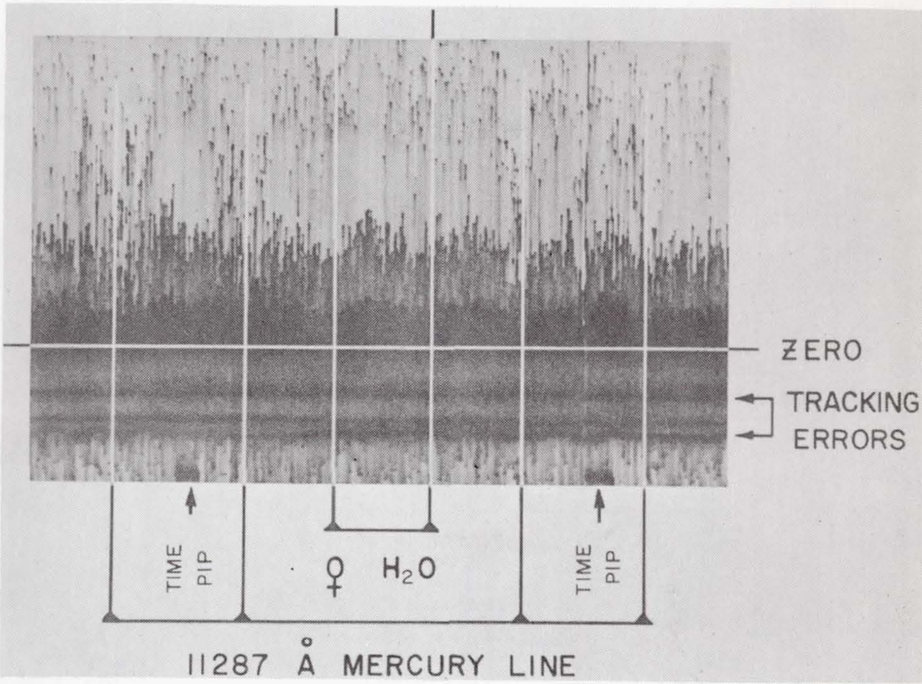

FIGURE 4

Double Scan Showing Absorption
By Water Vapor on Venus

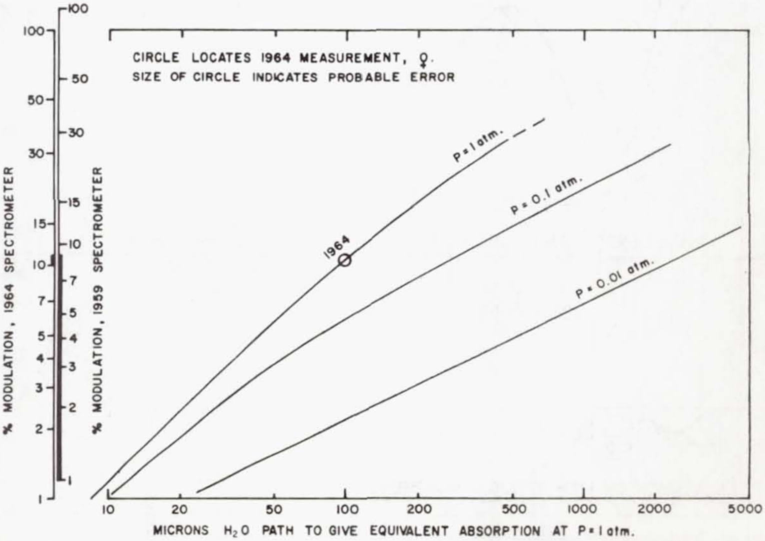

FIGURE 5

BAL-AST Spectrometer Calibration

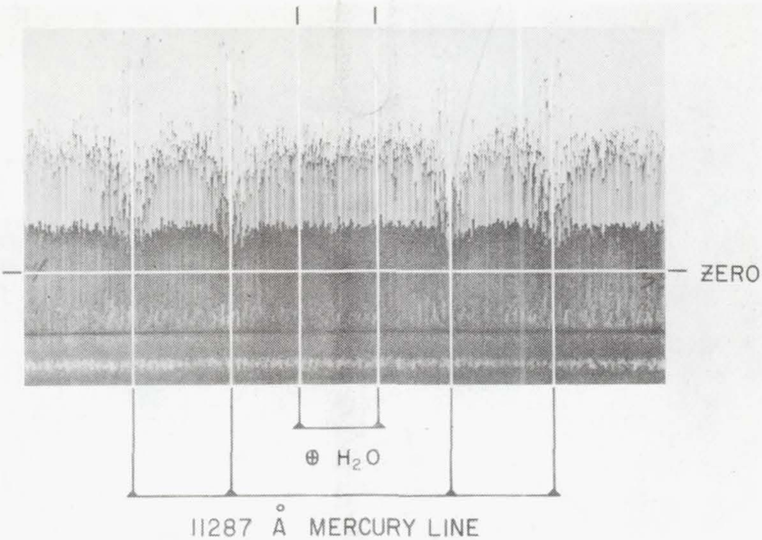

FIGURE 6

Terrestrial Water Above 26.5 km (Sunlight)

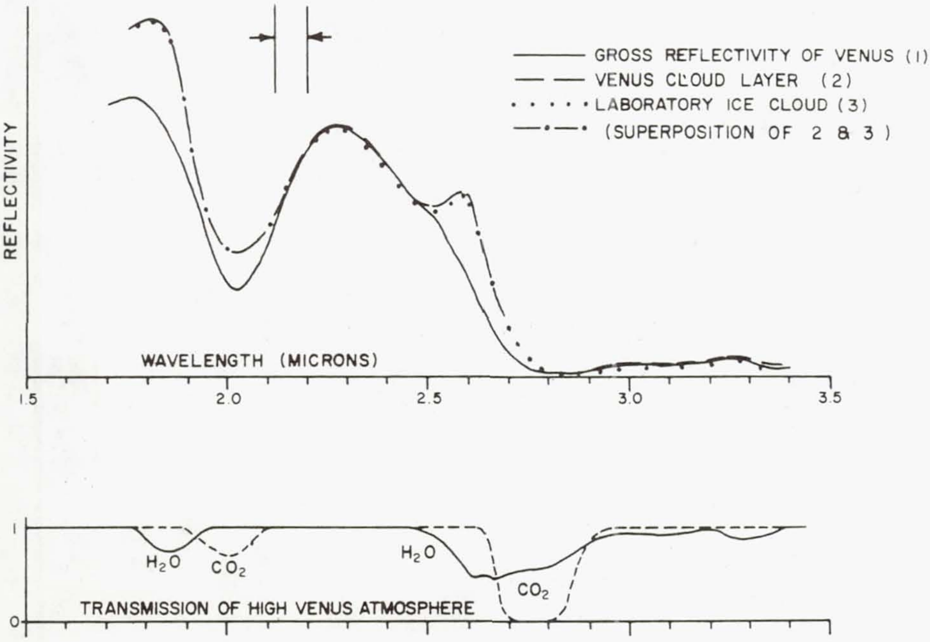

FIGURE 7

Reflection Spectrum of Venus Clouds Showing
Correction for Upper Venus Atmosphere

SOME ASPECTS OF THE CIRCULATION OF MARS

By

Conway Leory

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INTRODUCTION

The problem of the circulation of the Martian atmosphere is of great interest not only because of its close connection with a variety of other Martian problems, but also because of its implications for terrestrial meteorology. Both theory (Pedlosky, 1964; Phillips, 1963) and laboratory experiments on differentially heated rotating fluids (Fultz, 1961) indicate that the two important parameters determining dynamical similarity of the gross features of the flow are the Rossby number, $R_o = v/fL$, where V is a characteristic relative fluid velocity, f the Coriolis parameter, and L a characteristic scale of motion, and a static stability parameter, $\kappa = \delta s/c_p$, with δs the characteristic change in specific entropy over the depth of the circulation system, and c_p the constant pressure specific heat. The horizontal gradient in heating, primarily, determines V; L is either the planetary radius or a somewhat smaller scale associated with possible instabilities of the flow. For stability of the atmosphere with respect to small scale vertical convection, κ must be positive; it is determined jointly by the heating field and by the motion in a complex way: convective heat input near the ground and radiative cooling in the upper atmosphere tend to diminish κ; upward heat transport associated with large scale motions increases κ. The resulting value is determined by a balance between these processes, but for a given field of heating by radiation and small scale convection, κ increases as the intensity of the large scale circulation increases; hence it is ultimately related to the horizontal heating gradient. (For a discussion of the possible relationship between κ and the gross circulations of both Mars and the Earth as well as other aspects of the Mars problem, see Mintz, 1961).

The Earth and Mars have nearly equal rotation rates, and nearly equal axial tilts. Since the axial tilt is the most important factor in determining the differential heating, we can expect similarities in the differential heating on the two planets. Furthermore, since the lower atmosphere of both planets are heated by small scale convection near the ground and lose heat by radiation at higher levels, the relationship between κ and the circulation intensity should be similar. In some respects, the problem of the general circulation of Mars may be much more straightforward than that of the Earth. Water vapor, oceans, and clouds, all of which tremendously complicate the terrestrial problem, need not be considered on Mars. Consequently, the relationship between the external parameters: solar heat input, rotation rate, tilt, etc., and the resulting circulation should be simpler on Mars.

THE VERTICAL TEMPERATURE DISTRIBUTION

Since differential heating is an essential ingredient in the general circulation recipe, it will be necessary to consider its magnitude and distribution. It is helpful, however, to first review some of the main features of the expected vertical distribution of temperature as given by radiative equilibrium calculations. The temperature distribution derived from the radiative equilibrium hypothesis should give at least a rough qualitative picture of the real distribution. Furthermore, the calculations give some insight into the role played by radiation and small scale convection in determining the vertical temperature distribution, even though the large scale motions would modify this structure.

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Figure 1 shows a combination of separate calculations by Goody (1957) for the lower atmosphere and by Chamberlain (1962) for the upper atmosphere, as compared with the Earth. Several significant differences are evident: the Martian tropopause is higher, and the stratosphere is significantly colder. The temperature peak at the terrestrial 50 km level, which occurs as a result of absorption of solar radiation by ozone, does not appear on Mars. As we shall see, this fact may be significant for the problem of atmospheric tides. Chamberlain's calculations predict a deep temperature minimum near 130 km corresponding to the breakdown of Kirchhoff's Law for CO_2 to condense.

Despite these differences, the lower atmospheres of both planets are characterized by temperatures that decrease sharply with height. This is a consequence of the fact that most of the incoming solar energy in both cases is absorbed at the surface, rather than within the atmosphere. Radiation alone would produce a temperature discontinuity at the ground. It is assumed that small scale convection would smooth out the discontinuity and lead to an adiabatic lapse rate in the lower troposphere.

More recent detailed calculations for the lower atmosphere have been made by Prabhakara and Hogan (1965). They took into account the absorption of solar radiation as well as infrared emission by all of the important carbon dioxide bands, and also considered the possibility of absorption of solar radiation by small amounts of oxygen and ozone. A number of different possible combinations of surface pressure, carbon dioxide, and oxygen concentration were tried, but the results were relatively insensitive to reasonable changes in these parameters. Figure 2 shows their results in two of these cases. Except for different assumed surface temperatures there is a close resemblance between these temperature profiles and Goody's.

The depth of the troposphere cannot be determined on the basis of radiative equilibrium alone, since convection on both small and large scales helps to determine the tropopause height. These calculations suggest, however, that a troposphere some 2 or 3 times as deep as the Earth's is likely. By terrestrial analogy, we may expect that this tropospheric layer behaves as a single dynamical system, in the sense that the whole region would act as a heat engine. In this heat engine, solar energy received near the ground in equatorial regions or in the summer hemisphere increases the internal and potential energy. The latter are converted to kinetic energy of the horizontal winds, and in the process, heat is transported upward and horizontally to heat-sink regions. A small portion of the kinetic energy may be transported upward out of the troposphere and be reconverted to internal plus potential energy. The potential plus internal energy produced in this refrigerator-like process would be destroyed by radiation. The corresponding combination of heat engines and refrigerators in the Earth's atmosphere has been discussed in detail by Newell (1965). We shall confine our attention to the tropospheric heat engine on Mars.

DIURNAL HEATING AND TIDES

One class of large scale motions which might be important on Mars are thermally driven tides. Observations (Sinton and Strong, 1960) indicate that the diurnal surface temperature oscellation is on the order of 100°K. If this large amplitude is associated with a very large-amplitude diurnal component of vertical heat flux, significant tidal wind and pressure systems may be expected.

One way of estimating the diurnal component of the small scale vertical heat flux is to compare the observed surface temperature variations with those computed from a theory in which convective heat flux is taken into account. The amplitude and phase of the observed temperature wave can be matched by adjusting two parameters: the heat storage capacity of the ground and a heat exchange coefficient for the atmosphere (Leovy, 1965). A relatively simple theory that has been applied successfully to the Earth's atmosphere is that of Lonnqvist (1962, 1963). Lonnqvist assumes that heat exchange -- radiative, conductive, or convective -- between the ground and the atmosphere or space can be represented by Newton's Law of cooling. Such a relation is valid for convective transfer under conditions of forced convection and steady winds. The heat balance condition at the ground (z = 0) takes the form

$$k\left(\frac{dT}{dt}\right)_{z=0} + \sigma T^4_{z=0} - R_b + h_c[T_{z=0} - T_a] = S \quad , \qquad (1)$$

where the first term is heat flux into the ground; k is the soil's thermal conductivity to be determined. The second term is black-body emission from the ground. The third is back radiation from the atmosphere; the fourth is convective transfer between ground and atmosphere; h_c is a convective parameter to be determined, and T_a is a constant. S is the effective insolation. For an atmosphere without significant attenuation, the equation for S can be written

$$S = S_o(1 - A)(\sin \phi \sin \delta + \cos \phi \cos \delta \cos \omega t) \quad , \qquad (2)$$

where S_o is the solar constant at the distance of Mars, A the visible albedo, ϕ the latitude, δ the solar diclination, ω the diurnal frequency, and t is time. Of course S vanishes when the right side of (2) becomes negative. The black-body emission law can be linearized with only small error. The shape and amplitude of the computed temperature wave near the equator then depends on only two parameters:

$$A = (\pi \beta)^{-1}(\cos \phi \cos \delta) \; S_o(1 - A)$$

and

$$r = h/\beta \quad ,$$

where

$$\beta = (1/2 \; \rho c \; k\omega)^{1/2} \quad ,$$

ρ and c are the density and specific heat capacity of the top few centimeters of soil, and h is the sum of h_c and a constant obtained from line rizing the black-body emission law.

Figure 3 shows two surface temperature curves obtained in this way which fit Sinton and Strong's observations fairly well. The value of β is so low that, for any reasonable values of density and specific heat capacity, k must be of the order of 10^{-4} cal/cm sec °K or less. This is so low that soil particle size in the top few centimeters must be comparable to the mean free path of air molicules -- a few microns at the probably surface pressure of Mars.

The parameter h_c can be estimated from the two attempted fits to the data shown in Figure 3, and the corresponding most likely amplitude of the small scale convective heat flux is in the range 2--5 x 10^{-3} cal/cm^2 sec, or between 15 and 40 per cent of the peak insolation.

One can estimate the amplitude of thermal tides that this diurnally varying heat input would produce. A detailed study of possible Mars tides has been carried out by Craig (1964), who points out that the absence of any temperature maxima in the Martian stratosphere makes resonance amplification of the tides very unlikely. Calculation of tidal amplitudes should therefore not be very sensitive to the details of the vertical temperature structure. Craig computes the amplitude of the diurnal tide, which would arise from convective heating using a particularly simple model atmosphere and concludes that the maximum ratio of the diurnal tidal-pressure amplitude to the mean surface pressure should be about 1/600 or about the same as the corresponding quantity for the semidiurnal tide in the Earth's atmosphere. The convective heating assumed by Craig was only 1/3 to 1/8 as much as that estimated above, however. As long as the heating is confined to a thin layer near the surface -- a few kilometers deep, or less -- the exact distribution of the heat input with height is unimportant; only its amplitude affects the tidal amplitude. It follows that the Martian diurnal tide may be as much as 8 times as large as the terrestrial semidiurnal tide, and could be associated with near surface winds of 3 to 4 meters per second.

The diurnal tide is mainly excited by heating in thin layers; the semidiurnal tide is excited by heating through deeper layers. The absorption of solar radiation by CO_2 thus contributes to the semidiurnal tide. Comparison of the solar heating rates calculated by Prabhakara and Hogan with the corresponding quantity for the Earth's atmosphere suggests that this component of tidal forcing is comparable on the two planets. Qualitatively then, one would expect that the semidiurnal tidal amplitude on Mars would be comparable to that on Earth. More detailed calculations are not justified at this time because of the lack of detailed knowledge about the vertical temperature distribution. It is worth noting that very large tidal winds (greater than about 10 m/s) appear to be ruled out by the lack of indication for such oscillations in the observations of cloud drifts.

DIFFERENTIAL HEATING AT THE SOLSTICES

We come now to consideration of the atmospheric response to latitudinal and seasonal variations in heating. The results of a calculation by Mintz (1962) of the two most important heat balance components -- net incoming radiation and net outgoing radiation -- for the Martian southern hemisphere summer solistice are shown in Figure 4. A calculation of this kind is more straightforward for Mars than for the Earth because of the virtually complete absence of clouds and atmospheric water vapor on Mars. Furthermore, the difference between these two heat balance components, the net radiation excess, can be interpreted directly in terms of a heat transport requirement in the case of Mars. This is not possible for the Earth because of the large amounts of energy stored and transported by the oceans (seasonal storage of heat by the Martian soil must be negligible). Figure 5 shows the corresponding heat balance components for the northern hemisphere winter and summer of the Earth as computed by London (1957). It is ironic perhaps that no similar study for the southern hemisphere has been published yet, probably because of the uncertainties in cloud and water vapor distributions. Both planets show a net radiation excess in the summer and a deficit in the winter, but although such excesses and deficits can be interpreted as heat sources and sinks for the atmosphere on Mars, no such interpretation is possible for the Earth because of heat storage in the oceans.

Since the net radiation excess on Mars $Q(\phi, t)$ can be assumed to be balanced by atmospheric transport, one can write:

$$Q(\phi, t) = \frac{1}{a \cos \phi} \frac{\partial}{\partial \phi} \{\cos \phi \cdot 2\pi a^2 c_p \, p_o \, g^{-1} \cdot \overline{[v(T + gz \, c_p^{-1} + Lq \, c_p^{-1})]}\} \quad , \tag{3}$$

where ϕ is latitude, a is the planetary radius, p_o the surface pressure, and g is the acceleration of gravity. The bracketed quantity gives the advection of temperature T, potential energy gz, and latent heat energy Lq, where q is the specific humidity and L the latent heat of condensation. The advection is effected by the meridional wind component v. The bar indicates averaging over longitude, pressure, and time at a fixed season and latitude.

Noting that Lq can be neglected on Mars, this expression can be rewritten in the form,

$$H \equiv \overline{\{v[T + gz \, c_p^{-1}]\}} = [\cos \phi \cdot 2\pi a \, c_p \, p_o \, g^{-1}]^{-1} \cdot \int_{-\pi/2}^{Q} Q(\phi, t) \, d(\sin \phi) \quad . \tag{4}$$

for Mars and the Earth. This is done in Figure 6 for an assumed Mars surface pressure of 30 mb. Evidently the solstice's specific enthalpy transport requirement, which is a measure of the intensity of the thermally driven large scale circulation, is more than twice as large on Mars as on the Earth. This conclusion is strengthened when the effects of oceanic transport and storage, and latent heat transport are taken into account; it is also strengthened if the actual surface pressure on Mars is less than 30 mb. The values of H deduced in this way will be used to estimate the probably magnitudes of the winds which provide the heat transport.

GENERAL CIRCULATION AT THE SOLSTICES

When the Rossby number and the ratio of horizontal scale to planetary radius are both small (R_o, $L/a \ll 1$), a considerable simplification of the hydrodynamical problem is possible, provided also that the parameter,

$$\varepsilon = \frac{v^2}{gD\kappa} R_o^{-2} = \frac{f^2L^2}{gD\kappa} \quad ,$$

is of order unity or less. (D is the characteristic depth of the circulation system.) The simplification is known as the quasi-geostrophic theory. (See, for example, Charney and Stern, 1962; Phillips, 1963; Pedlosky, 1964.) These conditions are likely to be satisfied at middle and high latitudes on Mars, although not so well satisfied as on the Earth -- nevertheless, we shall make use of this theory in a qualitative discussion of the Mars winds. The quasi-geostrophic theory replaces the complete system of hydrodynamic and thermodynamic equations with a single equation and a single physical principle -- the conservation of potential vorticity, q, defined by

$$q = \frac{\partial^2\psi}{\partial x^2} + \frac{\partial^2\psi}{\partial y^2} + (D/L)^2 \cdot \frac{1}{\bar{\rho}} \frac{\partial}{\partial z} (\bar{\rho}\, \varepsilon\, \frac{\partial\psi}{\partial z}) \quad , \qquad (5)$$

where ψ is the stream function for the geostrophic wind, and $\bar{\rho}$ is the horizontally averaged density. The conservation of potential vorticity is expressed by the relation,

$$\frac{\partial q}{\partial t} + \frac{\partial\psi}{\partial x}\frac{\partial q}{\partial y} - \frac{\partial\psi}{\partial y}\frac{\partial q}{\partial x} = 0 \quad , \qquad (6)$$

where y is distance northward, x is distance eastward.

Charney and Stern have shown that, as a consequence of this equation, small disturbances in a basic zonal flow are necessarily stable if $\partial q/\partial y$ is of one sign and the latitudinal temperature gradient vanishes on the horizontal lower boundary. If the latitudinal gradient does not vanish, temperatures which fall toward the pole are destabilizing, and temperatures which rise toward the pole are stabilizing (Pedlosky, 1964). Because the gradient of f tends to dominate $\partial q/\partial y$, it is normally of one sign, and the effect of the temperature gradient at the ground seems to be the most important factor determining stability or instability. Results from other studies (Burger, 1962) suggest that small disturbances are always unstable if temperatures decrease toward the pole, the degree of instability increasing rapidly with the temperature gradient above a certain critical value of the latter. On the other hand, if temperatures increase toward the pole, small disturbances would fail to grow and in addition, any disturbances initiated at low levels by topography or local heating irregularities would be rapidly damped with height (Charney and Drazin, 1961). These theoretical differences between the two temperature gradient regimes are supported by experience in the terrestrial atmosphere. Thus we would expect that the summer hemisphere on Mars, in which daily mean temperature increase toward the pole, would be stable and nearly zonally symmetric. The heat transport would be accomplished by a mean meridional circulation. On the other hand, the circulation at middle and high latitudes in the winter hemisphere should be dominated by large-scale quasi-horizontal eddies which transport the heat. Momentum-balance considerations would then require westerly zonal winds in the poleward portion of the winter hemisphere and near the surface in the summer hemisphere, and easterly winds elsewhere.

We may now estimate the order of magnitude of the winds required to satisfy the specific enthalpy transport requirement. In the region where large scale eddies predominate, we have approximately

$$H \sim \overline{vT} \underset{\sim}{\sim} \sigma(v)\ \sigma(T)\ \ \rho(v, T) \quad , \qquad (7)$$

where $\sigma(v)$ and $\sigma(T)$ are standard deviations of meridional wind and temperature at fixed latitude and season at some representative midtropospheric height. The correlation coefficient $\rho(v, T)$ may be large for these eddies, and by terrestrial analogy, a value of $\rho(v, T) \sim 0.3$ appears reasonable. According to the quasi-geostrophic theory, wind and temperature are related by the thermal wind equation, so that

$$\sigma(T) \sim \frac{fL}{R} \left(\frac{D^*}{D}\right) \sigma(v) \quad , \qquad (8)$$

where R is the gas constant and D* is the scale height for Mars. Then

$$\sigma(v) \sim \{ [R/fL\rho(v, T)] (D/D^*) H\}^{1/2} \quad . \tag{9}$$

Assuming that the correlation coefficient and the scale, which are determined by the mechanics of the instability process,† are roughly comparable for Mars and Earth and that $D^* \sim D$, for middle latitudes we find that the ratio of standard deviations is

$$\sigma_m(v)/\sigma_e(v) \sim [(RHf^{-1})_m/(RHf^{-1})_e]^{1/2}$$

$$\sim (H_m/H_e)^{1/2} \gtrsim 2 \quad ,$$

where the subscripts m and e refer respectively to Mars and the Earth. The inequality is necessary since the values of H_e derived in Section 4 exceed the actual transport requirement. Equation 9 refers to the eddy velocity; the magnitude of the zonal wind velocity U is determined by a balance between loss of zonal momentum by vertical eddy stress, $K\, \partial U/\partial z$, where K is a vertical eddy stress coefficient, and production of zonal momentum by the large scale eddies. Assuming the kinematics of the unstable waves to be similar on the two planets, this production is proportional to $[\sigma(v)]^2$. Thus

$$[(K\, \partial U/\partial z)_m/ (K\, \partial U/\partial z)_e] \sim [(KU/D^*)_m/ (KU/D^*)_e]$$

$$\sim [\sigma_m(v)/ \sigma_e(v)]^2 \quad ,$$

and

$$(U_m/U_e) \sim [(KD^{*-1})_e/(KD^{*-1})_m] \cdot [H_m/H_e] \sim 6 \quad , \tag{10}$$

assuming $K_m \sim K_e$. Thus zonal winds of several hundred meters per second are possible at one-scale height above the Martian surface. The surface zonal wind ratio, on the other hand, would be comparable only to $(H_m/H_e)^{1/2}$

For the mean meridional circulation regime of the winter hemisphere, it is somewhat more difficult to estimate velocities. It follows from Equation 4 that

$$H \sim v(\gamma - \gamma_a)\, D_e \quad , \tag{11}$$

where v is the mean meridional wind component, $(\gamma - \gamma_a)$ is the difference between the actual and the adiabatic lapse rates (the adiabatic lapse rate is g/c_p), and D_e is the depth of the Ekman friction layer,

$$D_e \sim \pi(2Kf^{-1})^{1/2} \tag{12}$$

(Taylor 1915). Near the equator D_e must be replaced with $D \sim D^*$. D_e and $(\gamma - \gamma_a)$ are difficult to estimate, but as an illustration of the magnitudes involved, we may take the reasonable values $H \sim 50$ m/s deg, $(\gamma - \gamma_a) \sim 2°$/km, $D_e \sim 2$km, giving $V \sim 20$ m/s. This is a substantial mean meridional wind, and would be associated with a still larger zonal component.

Figure 7 illustrates these ideas schematically. The winter hemisphere contains strong zonal west winds that increase with height, except for a shallow belt of easterlies near the ground in low latitudes. The summer hemisphere is dominated by easterlies increasing with height -- except for a relatively shallow belt of surface westerlies. The meridional circualtion associated with the eddy regime has descending motion equatorward of the west wind maximum and ascending motion poleward of the maximum. This circulation is a consequence of the production of zonal momentum by the eddies. In summer, a shallow poleward flow takes place beneath a deep, gradually ascending, return flow.

Although the Mariner 4 results (Kliore <u>et al</u>., 1965) would alter some of the quantitative estimates given above through the surface pressure that enters the expression for H_m, the main features shown in Figure 7 still appear to be very likely.

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† *Actually L will be a function of κ; see Mintz (1962).*

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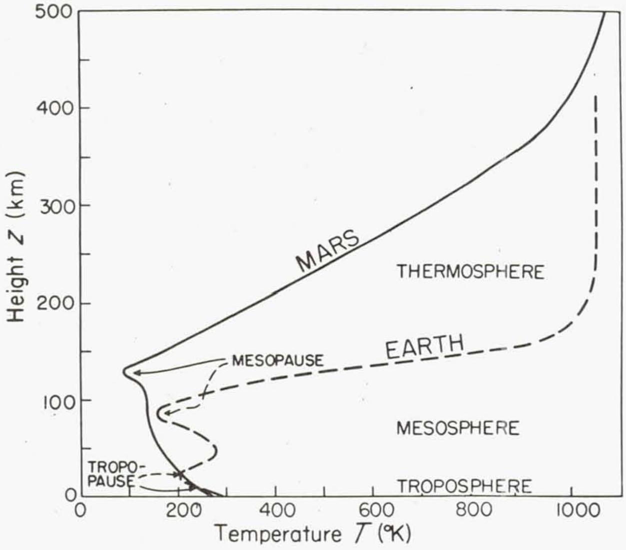

FIGURE 1

Vertical temperature distribution on Mars as calculated by Goody and by Chamberlain compared with the vertical distribution in the Earth's atmosphere (from Chamberlain 1962).

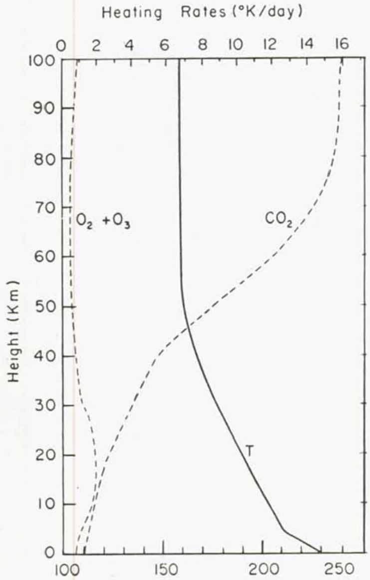

FIGURE 2a

Surface Pressure 10 mb, 44% CO_2, 0.4% O_2

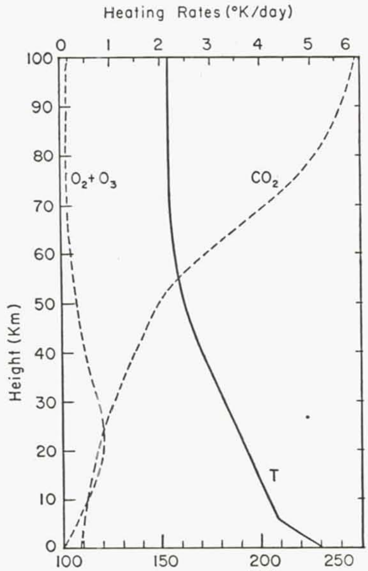

FIGURE 2b

Surface Pressure 30 mb, 9% CO_2, 0.106% O_2

Vertical temperature distributions for the lower atmosphere of Mars (from Prabhakara and Hogan 1965).

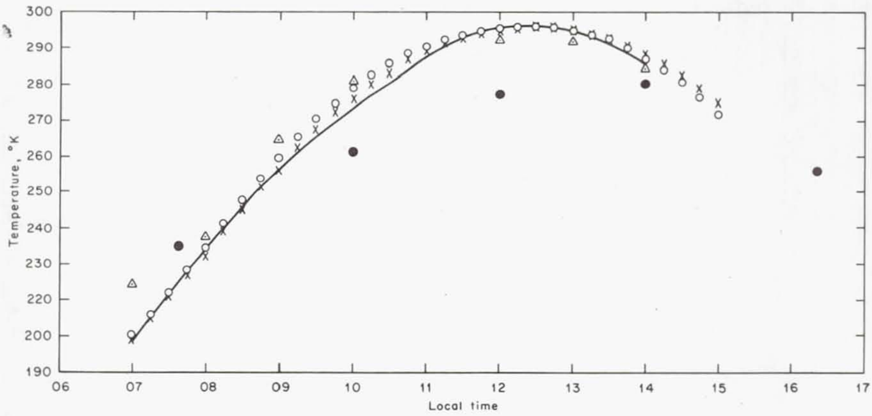

FIGURE 3

Theoretical fit of Sinton and Strong's observations. Solid line is the average of their four best temperature curves; (x) are the fit for A = 236.5, r = 6, open circles are for A = 379, r = 10. The triangles are for data taken predominantly on dark areas. The solid circles represent the average of earlier observations by Coblentz and Lampland as presented by Gifford (1956).

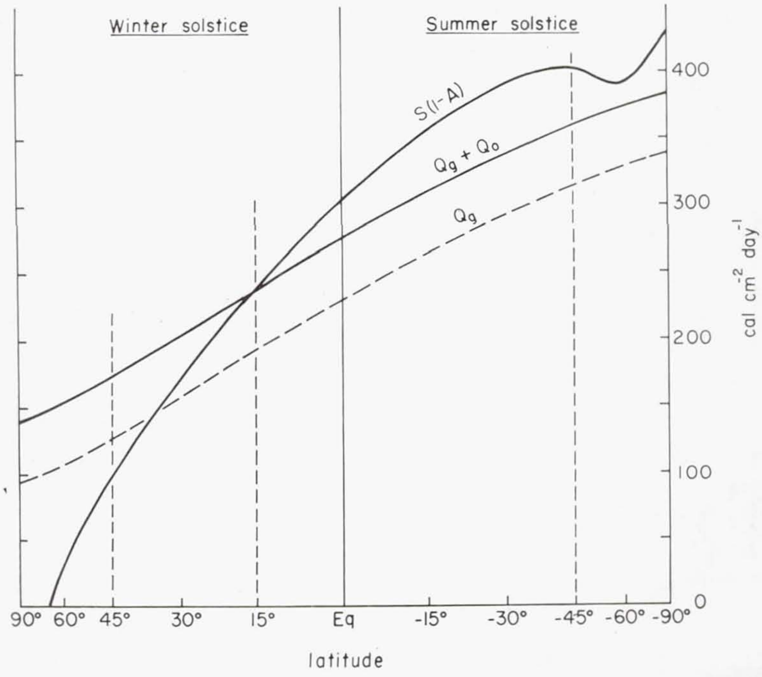

FIGURE 4

Martian heat balance components. S(1 - A) is the net incoming radiation; Q_g is the radiation to space from the surface, $Q_g + Q_o$ is the total radiation to space from the surface and the atmosphere (from Mintz 1962).

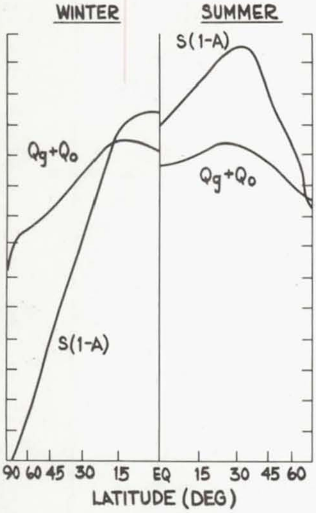

FIGURE 5

London's data for the net incoming and net outgoing radiation on the Earth.

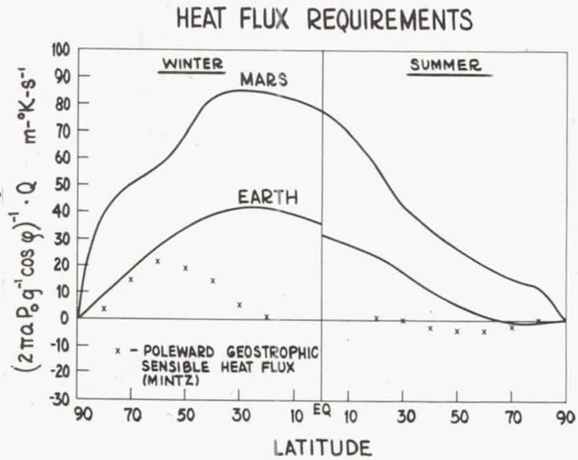

FIGURE 6

The specific enthalpy flux requirements for the Earth and Mars. The poleward sensible heat flux calculated by Mintz (1954) and indicated in the diagram gives only the contribution from eddies. Any contribution from mean meridional circulations is not included in his data.

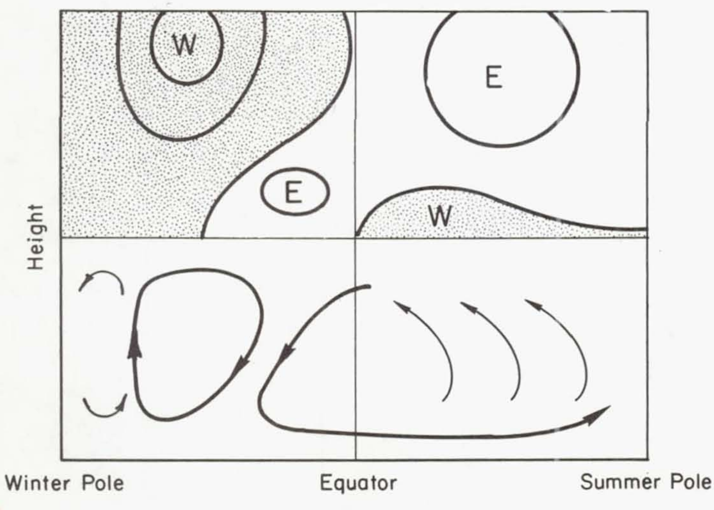

FIGURE 7

Schematic diagram of solstice circulation on Mars. Top half is zonal circulation, bottom half is meridional circulation.

THE INTERIORS OF MARS AND VENUS

By

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INTRODUCTION

Observations of Mars and Venus played an important role in establishing the fundamental laws of dynamics in the seventeenth century. The orbit of Mars has a relatively large eccentricity (.093). This fortunate circumstance in a planet with considerable mean motion enabled Kepler to establish that planetary orbits are ellipses with a focus at the Sun's center. Galileo's discovery with his telescope of the phases of Venus provided important support for the Copernican theory. Today the exploration of the inner planets provide tests for the concepts built up in the last half century in geophysics. Theories that the geomagnetic field results from a dynamo process in the earth's fluid core, that convection currents in the earth's plastic mantle explain mountain building, continental drift and the low harmonics of the earth's gravity field, that high pressure modifications in silicate structures determine the density distribution and theories of accretion, all require testing by observations of other planets. Negative results can be significant.

The essential difficulty of the physics of the interiors of the inner planets is that theoretical physics is not of great help. Eddington's classical studies of the internal constitutions of stars was so successful because it was clear that the gas laws would apply to the state of matter in their interiors. The major planets largely consist of hydrogen, and the quantum mechanics of hydrogen molecules is simple enough for solution and valuable predictions can be made about the interiors of Jupiter and Saturn. For the earth's mantle the application of theory to the complex silicate molecules does not yet yield many useful results. However seismological studies, combined with knowledge of the earth's mean density and its moment of inertia, give values for the density and elastic constants of silicates and iron up to pressures of 1 million bars. With this information models of Venus and Mars can be computed.

CHARACTERISTICS OF MARS AND VENUS

We first review the basic information available about the two planets. The mass of Mars is accurately known from the periods and semi major axes of the orbits of its satellities Deimos and Phobos. From both, the mass of Mars in terms of the earth's mass is 0.1076 or 3,093,000 ± 3000 reciprocal solar masses (Wilkins 1964). The mass of Venus had formerly been determined from the small perturbations made on the earth's orbit but a more accurate mass (0.8136) has recently been deduced from the path of Mariner II, as it passed the planet. The radii of Venus and Mars determined optically are 0.973 and 0.520 respectively of that of the earth. There has been difficulty in determining the radius of Mars as different observers have observed different values but the value quoted above is taken in the infrared and is likely to refer to the solid surface. These values now give mean densities of Venus and Mars to be 4.88 and 4.24 respectively to be compared with 5.52 for the earth.

While we cannot determine the moment of inertia of Venus, that of Mars can be found from the precession of its satellites. Let C be the polar moment of inertia and A and B that about perpendicular axes in its equatorial plane.

MacCullogh's relation gives the potential U at field point (r, θ, ϕ)

$$U = \frac{GM}{r} + \frac{G(A + B + C - 3I)}{2r^3} \qquad (1)$$

where I is the moment of inertia about the line joining the center of the mass to the field point.

As

$$I = C\cos^2\theta + A\sin^2\theta\cos^2\phi + B\sin^2\theta\sin^2\phi \quad , \tag{2}$$

$$U = \frac{GM}{r} + \frac{G[C - A](1 - 3\cos^2\theta)}{2r^3} \tag{3}$$

supposing A = B.

If the surface of Mars is an equipotential and (r, θ) is a point on this surface, then

$$U + {}^1/_2\,\omega^2 r^2 \sin^2\theta = \text{constant} \tag{4}$$

If it is not an equipotential, one may be constructed and in either case is approximately an ellipsoid of revolution about the polar axis of ellipticity ε. Thus

$$\frac{x^2}{a^2} + \frac{y^2}{a^2} + \frac{z^2}{a^2(1-\varepsilon)^2} = 1 \tag{5}$$

where (x, y, z) is the cartesian coordinates of the point on the equipotential surface, z being the polar axis, and a is the equatorial radius.

Substituting (5) and (4) in (3) to the first order of small quantities

$$U = \frac{GM}{r} + \frac{GMa^2}{r^3}(\varepsilon - \frac{1}{2}\Phi)\ (\frac{1}{3} - \cos^2\theta)$$

where Φ is the ratio of the centrifugal to the gravitational forces at the equator and ε is known as the dynamical ellipticity.

By Gauss' device of supposing the satellite is replaced by a ring of equal mass occupying the circular orbit, we can compute the torque on an element dm of the ring at longitude

$$dm\sin^2\lambda \left.\frac{dU}{d\theta}\right|_{r=R} = \frac{GM\,dm\,a^2(\varepsilon - \frac{1}{2}\Phi)}{R^3}\sin 2\theta\sin^2\lambda$$

where the radius of the orbit is R.

This is equal when summed, to the rate of change of angular momentum of the satellite ring = $mR^2(2\pi/T)(2\pi\sin\theta)/t$ where T and t are the orbital and precessional periods of the satellites respectively and m is the satellite mass.

Thus, if θ is small,

$$t = \frac{R^2}{a^2}\ \frac{T}{(\varepsilon - \frac{1}{2}\Phi)}$$

Woolard (1944) found that the rates of precession of the nodes of the orbits of Phobos and Deimos are 158° .484, and 6°.27950 per tropical year, when allowance is made for the Sun's contribution, and if a = 0.520, then the data gives $\varepsilon = 0.0052$ and 0.0051 respectively. Wilkins (1964 has substantially confirmed these results by new computations of the orbits and has shown that the rate of rotation of the pericentre of the orbit of Phobus is equal to that of the node of its orbit, as theory, to a first order, predicts (Brouwer and Clemence 1961). Thus the dynamical ellipticity is very accurately determined, and ε/Φ is found to be 1.22.

Were Mars known to be in hydrostatic equilibrium we could determine C as follows. Each equipotential surface within Mars is an equal density and pressure surface (of varying ellipticity). An approximate treatment by Radau and Darwin gives

$$\frac{C}{Ma^2} = \frac{2}{3}\left[1 - \frac{2}{5}\left(\frac{5}{2}\frac{\Phi}{\varepsilon} - 1\right)^{1/2}\right] \tag{6}$$

Thus

$$C = (.984)\ (\frac{2}{5}Ma^2)\ ; \tag{7}$$

thus Mars is close to being a body of uniform density.

It is, however, doubtful if this procedure is valid. The surface ellipticity has many times been determined by optical methods in yellow light. De Vaucouleurs (1964) reviews the best and gives 9".315 ± ".010 and 9".415 ± ".02 as the polar and equatorial diameters at unit distance respectively. The ellipticity is then 0.0105 ± 0.0005 (p.e.). In red light and from the motion of surface markings, the smaller

values 9".19 ± 0.03 and 9".28 ± 0.03 are obtained and may refer to the solid surface. In either case the ellipticity determined is about twice that found by dynamical methods but as it is likely to have a larger error than the dynamical ellipticity, it has been disregarded. It has been explained away due to distortion in Mars atmosphere, or as due to an equatorial belt of high land - hypotheses less attractive since the Mariner IV photographs. The presistant disregard of the optical ellipticity result from the awkward problem which arises if it is substituted in (6); an imaginary number is obtained. There is, however, strong reason to ascribe the discrepancy to the departure of Mars from hydrostatic equilibrium. We may then conclude that the equipotential surface in the atmosphere of Mars tangential at the equator to the surface of Mars is elevated 16 km above the poles. The measurements of the thickness of the atmosphere during the occultation of radio signals from Mariner IV were at 55°N and 60°S. Measurements at two different latitudes would have enabled this important question to be settled. The astronomical methods give an average over the disc of 83 mb (Dollfus) c.f. 7 mb from Mariner IV. The discrepancy is one which it is important to test critically; the astronomical method should be applied to the equatorial and polar regions.

The existence of this bulge can be explained by postulating extreme rigidity in the planet, its origin being bound up with the planet's formation, as it is inconceivable that a recent event could cause it. The bulge represents a strain of 1/200 and its maintenance requires a stress difference of 1000 bars or 10^9 dyne/cm^2. Classical elasticity and the existence of a finite stress below which flow did not occur even over long times, were hypotheses about the mechanics of solids which were long in the discussion of geophysical problems. The evidence for continental drift has caused a more sophisticated view to be taken - one more in harmony with the modern theories of solids (Runcorn 1962). Flow may be assumed to occur even under very small stresses over times of the order of the geological scale at depths in the earth below a few tens of kilometers where the temperatures are elevated. It is useful, though not strictly correct, to use a viscosity. In order that Mars be rigid enough to maintain the bulge since its formation 4×10^9 years ago (about 10^{17} secs.), its internal viscosity must be greater than 10^{28} poise.

Gordon (1965) has argued that diffusion of atoms through grains is the important creep mechanism in the polycrystalline earth's mantle at low stresses, and one which is bound to occur even if other creep processes are absent. The atomic migration is driven by the applied stresses; its vacancy sources and sinks being the grain boundaries. Thermal activation causes the diffusion and thus in the Earth's mantle, the viscosity so calculated rises by more than six orders of magnitude in the first one or two hundred kilometers from the surface due to the high geothermal gradient near the surface. Gordon finds that pressure causes an increase of viscosity below about 500 km depth, but as the pressure then exceeds that of the center of Mars, we can conclude that viscosities lower than 10^{28} poise are to be expected in Mars, except near the surface, and unless the temperatures in its interior were everywhere less than about 1000°C.

The possibility that convection within Mars is a cause of the distortion of the surface has not hitherto been considered. It is easy to see that this must be a second harmonic symmetrical about the axis of rotation with hotter, less dense material rising around the equator and the colder falling at the poles. Were no motion taking place, the systematic difference in temperature necessary to cause the observed bulge would be 500°C, taking 3×10^{-5} per °C as the volume coefficient of expansion of olivine. The process of convection would increase the required temperature difference, and this seems remarkably large. However, even if such convection exists, theory and experiment alike suggest that a second harmonic convection would only occur if the core of Mars was no greater than one third of the radius of the outer boundary of the convecting shell; i.e., perhaps 50-200 km below the surface.

If it were proved that the optical ellipticity is an observational effect, then Eq. (7) applies and again the radius of the core can be no greater than a few hundred kilometers.

In contrast to Jupiter and Saturn, Mars emits no non-thermal radiation and thus neither a radiation belt nor the associated planetary magnetic field is assumed to exist. This need not be held to support the hypothesis of no core in Mars; but merely, that all the conditions for the spontaneous generation of a field by dynamo action are not present. These hypotheses are that motions in a fluid core provide energy, derived presumably from energy released by radioactivity, to maintain the field. The core of Mars even if it exists may not be fluid.

Further the core must be of a certain size, other physical constants being equal, if the generation of new lines of magnetic force are to predominate over the natural decay of a magnetic field in a conductor of finite conductivity. A liquid core of such a small radius as that which we have been considering is not likely to be a dynamo (Runcorn 1965).

A further piece of evidence points to the core of Mars being small or non-existent or solid. The liquid cores of the Earth and Jupiter do not rotate at exactly the same speed as the mantles; there is no reason why they should - the viscous forces in spheres of such size are negligible. Only the weak electrical conductivity in their mantles - the result of semi-conduction processes - enables coupling to occur between the mantle and core. If the planetary magnetic field were constant and rotating with the core, any relative rotation of core and mantle would die away because of eddy current losses, providing, as is now the case with the Earth and Jupiter, the magnetic and rotational axes are not the same. However, in the Earth the magnetic field varies due to turbulance in the core. We may predict that a phenomenon similar to the geomagnetic secular variation will occur in the length of the Earth's day. A similar rather irregular change in the rotation of Jupiter occurs, with periods of about 50 years. This follows from Hide's theory of the red spot, that it is an atmospheric column anchored to a major surface feature on the jovian mantle (perhaps a large meteor crater). On the other hand, Ashbrook (1953) has shown that the period of rotation of Mars has remained remarkedly constant to a thousandth of a second since the eighteen century. This is an order of magnitude more constant than the rate of rotation of the Earth and again argues that the moment of inertia of Mars core is proportional much less than the Earth's and to the absence of significant electromagnetic coupling between the mantle and core.

INTERIOR OF THE TERRESTRIAL PLANETS

Further investigation of the interior of the terrestrial planets may be made by use of data derived from seismology. Travel times of the P and S waves from earthquakes have enabled the velocities v_p and v_s respectively of these sound and shear waves to be determined at every radial distance r.

$$v_p = [(k + \frac{4}{3}\mu)/\rho]^{1/2} \qquad \text{and} \qquad v_s = (\mu/\rho)^{1/2} \qquad (1)$$

where k is the bulk modulus or incompressibility, and μ is the shear modulus and ρ the density. In any region of the earth when the chemical composition is constant and the temperature gradient adiabatic

$$k = \frac{dp}{d\rho} \qquad (2)$$

where p is the pressure at radius r.
Thus

$$v_p^2 - \frac{4}{3}v_s^2 = \frac{dp}{d\rho} \qquad (3)$$

Hydrostatic equilibrium is very closely satisfied in the earth (say to $1/10^5$).
Thus

$$\frac{dp}{dr} = -g\rho \quad , \qquad (4)$$

where g is the gravitational acceleration at radius r.

If the earth were chemically uniform, the known surface gravity and density would enable Eqs. (3) and (4) to be integrated throughout. In order, however, to obtain a density radius curve which allows the total mass and moment of inertia of the Earth to be obtained, two discontinuities have to be taken into account - the core mantle interface with a sharp density discontinuity and a less pronounced discontinuity around 500-900 km depth. Bullen has thus determined the variation of the Earth's density with radius. It has been suggested that the core is a phase change - the silicate of the mantle becoming metallic. The older suggestion that it consists of liquid iron-nickel now seems to have received new support from high pressure experiments by Takahaski and Bassett (1965) who show, above 130 kilobars, iron goes over into a hexagonal close packed phase (ε iron). They argue that this is the phase present in the core, and show that above 230 kilobars, the nickel iron alloy is lighter than pure iron. In the upper mantle the known rapid rise of temperature with depth and the likely phases changes of olivine from rhombic to cubic form make it doubtful if the Equation (2) can be used above a depth of 700 km. Using Bullen's data it has been noticed that in the lower mantle and core

$$k = a + bp$$

where b = 3.5 and the constant a in the two cases is slightly different. Lyttleton (1965) has drawn attention to the value of this linear relation in constructing planetary models. He also fits the upper 700 km of the Earth's mantle by a similar law with different a and b, but the data is not so well adapted to this representation and I doubt whether its relation in this region has the physical significance which it has in the core and lower mantle. It seems likely that no phase changes occur at higher pressures. In this region the simplest type of potential function (U) between pairs of ions may be taken as,

$$U = \frac{e^2}{r} - \frac{A}{r^n} \qquad (5)$$

where e is the ionic change, r the distance apart of the ions and n the exponent in the Lennard Jones law of repulsion between closed shells. Then r_o, the cell size, is given by making U a minimum.

Thus

$$A = \frac{e^2}{n} r_o^{(n-1)} ,$$

r_o being related to the density at zero pressure. Increases pressure adds a term proportional to p to Eq. (5) and we find b = n/2, a reasonable value on theoretical grounds.

We now can feel confident about applying these relations to the interiors of Mars and Venus, which Lyttleton (1965) has done. On the hypothesis that the Earth's core is a phase change, Lyttleton is able to exclude a core for Mars, as its central pressure is not high enough to cause a phase change. He then fits the mass of Mars to a model consisting of two regions corresponding to the terrestrial upper and lower mantles.

MacDonald (1962), regarding the Earth's core as iron, fits an iron core and a silicate mantle, making in one case the assumption of a silicate phase change at 10^5 bars. He finds, for a value of the ellipticity 0.005. The mass of the core is 0.01 of Mars mass in the phase transition model and 0.093 with no phase transition.

Sharpless (1945) argued that a secular acceleration in the longitude of Phobos amounting to 8×10^{-12} in one period existed. This would probably have had to be explained in terms of a non-elastic behavior of Mars, there being no ocean as on the Earth in which tidal friction could occur. However, recently Wilkins (1964) has reanalyzed the observational data from 1877-1929 and finds no significant acceleration. This new work removes an awkward difficulty. Even if creep occurs in a body over millions of years it is probable that its behavior in response to the stresses raised by the tidal pull of Phobos (of a few hours period) is purely elastic.

The difficulty in both MacDonald's and Lyttleton's method lies in the interpretation of the earth's upper mantle. Birch (1951) showed that this was the region where chemical inhomogenity or a phase change was possible. Phase changes in silicates are now established experimentally around the pressures of 10^5 kilobars. Furthermore in part of the earth the large temperature gradients may appreciably influence the variations of k.

It is perhaps when we try to determine the temperature distribution within the terrestrial planets that the greatest doubts arise. Urey (1952) and MacDonald (1962) have considered the thermal histories of Mars and Venus, on the assumption that radioactivity is uniformly distributed as that of chrondritic meteroites and the heat transfer is by conduction alone. Radiation recently has been shown to be of importance in the deep interior of the earth as it is proportional to the fourth power of temperature. The result of calculations gives a rapid rise of temperature to the depths about one quarter the radius of the planet and the temperature gradient then rapidly diminishes with depth - cooling extending appreciably only to a few hundred kilometers even over 10^9 years. If convection is postulated, the temperature gradient at depth is smaller and equal to its adiabatic gradient. Again in the first few hundred kilometers the gradient is steep because heat transported by convection in the deep interior must be transferred by conduction through the rigid shell. Because creep depends so sharply on temperature, a good model of any terrestrial planet consists of a rigid shell and a plastic interior rather sharply divided. It should not be concluded then that because the surface of Mars neither shows evidence of folded mountains nor of volcanism that internal motions or convection are not occurring. On these views tectonic phenomena result from the stresses produced by convection on the rigid crust but they must be sufficient to cause fracture. Runcorn (1965) shows that this is possible in the Earth, but if convection was, say, an order of magnitude less rapid, tectonic features would not occur.

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THE SYSTEMATIC INVESTIGATION OF THE METEOROLOGY OF MARS

By

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INTRODUCTION

The original letter of invitation for this talk recommended the title, "A Future Automated Weather Station." Since my preoccupation for the past six years has been with the development and launch of weather satellites around Earth, my initial interpretation of the letter of invitation was that what was required was a look into the future of these automatic weather stations that we are developing.

I could have proceeded along these lines and first summarized for you the magnificent achievements of the 10 successfully launched TIROS satellites and the one Nimbus satellite. I could have continued with our plans with the Weather Bureau for implementing TOS, the operational weather satellite system based on TIROS, and with our plans for the forthcoming Nimbus satellite.

I could have commented on our programs for the synchronous meteorological satellite which could provide continuous weather observations from space.

I could have described this country's plans for the World Weather System based on the weather satellite.

And indeed I am prepared to do so.

And I would have been quite responsive to the suggested title for this talk.

But then I stopped to reflect on the theme of this conference, "The Exploration of Mars and Venus," and I posed the question -- how does all this fit into the problem of unmanned and manned planetary exploration? I concluded that it doesn't, and so I was ready to decline the invitation. But then I considered the more general problem -- what is the relation of weather to the exploration of the planets -- the manned exploration. And gradually some ideas formed in my mind.

My objective today is to share these preliminary ideas with you. The ideas require considerable evaluation and study -- they are bones that need to have flesh put on them -- they are ideas that need to be quantified. But they are some ideas that I have considered on the "Systematic Investigation of the Meteorology of Mars."

THE MARTIAN EXPLORATION OF EARTH--PROJECT MEL

A little over a month ago, on July 14, Mariner IV passed within 7,000 miles of the planet Mars and, as has been discussed with you in this Conference, Mariner IV relayed back to us some very exciting information about that planet. I am sure you have all seen the remarkable pictures of Mars that were taken by Mariner IV. Figure 1. is one of the more striking pictures of the series. This picture, and the others in the series, will be studied for many months and years in order that all the scientific information may be extracted from them.

This morning I would like to share with you some additional information reputedly collected during the flight of Mariner IV; information which is so astonishing that one can hardly believe it. How this information was relayed, decoded and reconstructed, and how it was brought to my attention is a story in itself. Today we shall have time to discuss only the actual message itself. And it was a message, indeed!

When decoded, the message was revealed to be a narrative by the Martians of their space program at some unspecified time and, in particular, of their program for Manned Earth Landing--which they appropriately called Project MEL. The date of the activity is obscure. The record refers to a time relative to the simultaneous explosions of several super-nova--an event unfamiliar in our recorded history. And so, in essence, I shall reveal to you what might be considered the contents of a message which has been in continous transmission for several thousand years and which describes space activity of Mars that took place that many years ago.

The Astronomical Description of Earth

Martian astronomers had been interested in Earth for many years but had difficulty in studying it due to the bluish haze that surrounds the planet. They had long determined that the white areas--the permanent, the semipermanent, and those that migrated across the disc of the planet--consisted of hydrogen and oxygen in combination as H_2O. This compound appeared in several forms, as a vapor, as a liquid and as a solid, and there was a constant change from one phase to another due to temperature changes. Surface planetary temperatures varied very little, about $100^{\circ}K$ from one absolute extreme to the other. This absolute range was comparable to the diurnal surface temperature variation on Mars. Surface pressure was determined to be very high, probably exceeding 800 millibars or more. Earth beings, if they existed, would have to be highly adaptable to moisture changes--but not so much to temperature changes. With the discovery of the corrosive element O_2 in great abundance on Earth, it was generally agreed that life on Earth was highly doubtful--at any rate, so far as life was known on Mars.

The Earth Orbiter

These early astronomical findings could provide but rough clues on the nature of and possible life on Earth. With the development of suitable vehicle propulsion and information transmission techniques, the Martian space program developed and grew. After the successful launch of simple artificial satellites around Mars and landings on Deimos and Phobos, the natural moons of Mars, the Martian exploration turned to the nearby planet, Earth. The first Earth orbiter passed very close to the Earth at a distance of only 2,000 km. and transmitted back pictures of the planet. Figure 2. is one of the early pictures so transmitted. The details of the white areas were clearly revealed, as was their global organization. The existence of large hemispheric patterns as well as small scale patterns led to the conclusion that atmospheric motions on Earth were of many scales. The atmospheric vortex was the predominant and most spectacular formation. Little detail could be seen in the dark areas. These early pictures of Earth showed no cultural imprints, no evidences of cities, nor of canals, to lend support to the theory that there was life on Earth. The Martians then turned to the problem of landing a Martian on Earth as precursor to the overall colonization of the planet.

The Earth Capsule

It was evident that too much uncertainty existed concerning the nature of the Earth's atmosphere and that it was not possible to agree on the required design specifications for a manned spaceship to send to Earth. Consequently, it was decided, as a preliminary measure, to develop a smaller Earth capsule. This capsule was to be ejected from an Earth orbiter and would take appropriate measurements during its descent to Earth.

Plans were for two such capsules--one the White Area Probe, the WAP; and the second the Dark Area Probe, the DAP. By probing the conditions in both white and dark areas, it was expected to establish the extremes of conditions on Earth. The following were the results of the successful launch of probes to Earth from Mars:

1. The White Area Probe (WAP) Results

Figure 3. shows the data provided by WAP during its descent to the surface of the Earth. You will note the temperature profile reached a minimum at some altitude during descent and then increased towards the surface of the Earth. It was not clear whether the final decrease was real or whether it was a measurement and/or transmission error. The surface temperature was about $5^{\circ}C$, somewhat warm compared to Martian climate. There was a high concentration of moisture during descent and solid precipitation fell in the white areas at a rate of about .05" per hour. Strong winds of over 100 knots were encountered aloft, but on the surface they were no greater than about 20 knots.

2. The Dark Area Probe (DAP) Results

The DAP results given in Figure 4. showed, as expected, some markedly different characteristics. The upper air temperature was generally isothermal and the surface temperature was much lower, about $-30^{\circ}C$, thus quite comparable to the Martian climate. There was little wind at the surface and aloft. There was very little moisture in the dark area and no solid precipitation could be measured.

(I should mention, at this point, that, in reviewing these profiles, we came to the conclusion that the WAP traversed a cold frontal region in the middle latitudes, while the DAP landed in a Siberian anticyclone in the winter.)

But now, continuing with the narrative as provided by the Mariner IV relay of the transmission from Mars...

The White Area Manned Earth Lander - WAMEL

Having data on these extremes available, the Martian engineers designed Manned Earth Landers to withstand, with ease, anticipated weather related hazards. The first MEL was launched to land in a white area. It was designated WAMEL--the White Area Manned Earth Lander. The launch was highly successful, the spacecraft midcourse correction was perfect, and the descent was as planned. Unfortunately, Martian scientists received only a brief transmission from WAMEL. The transmission indicated that the lander had run into serious problems in the atmospheric conditions experienced by the Martian astronauts in their descent into a white area. The surface temperature was not $5^{\circ}C$ but $20^{\circ}C$, the surface winds were not 20 knots but about 125 knots, and the precipitation was liquid, not at the rate of .05" per hour but many inches per hour (over 5"). The astronauts reported considerable buffeting in the lower part of their descent due to wind, and severe flooding of the spacecraft by water. After this short transmission, nothing further was heard from WAMEL. It is our estimate that the WAMEL had the misfortune of entering a U. S. East Coast hurricane.

The Dark Area Manned Earth Lander - DAMEL

The second MEL aimed at a dark area--the Dark Area Manned Earth Lander (DAMEL)--was successfully launched and it, too, descended to the surface of the Earth without mishap but, here again, there were problems. The Martian astronaut reported that, on egress from his spaceship, the temperature was not the expected $-30^{\circ}C$ but an unbearable $45^{\circ}C$. DAMEL thus ended catastrophically also. It is our estimate that DAMEL descended in a desert region, either in Death Valley or the Sahara.

The Martian record stops here. At this time, we do not know whether additional attempts were made to land on Earth.

WEATHER ON EARTH

Observations

As I stated earlier, this story of the receipt of a message from Mars by means of Mariner IV, describing an earlier Martian exploration of Earth, is indeed unbelievable. Frankly, my personal assessment is that it is pure fiction.

Whether or not one wishes to put any credence in this account, one can nevertheless learn an important lesson from this inversion of the foles of Earth and Mars. As shown on figure 5. there were significant differences in the probe and lander measurements for supposedly similar kinds of areas. Thus, one must be very careful not to use point observations provided by a single (or even several) probe(s) as the basis for defining the range of conditions to be expected. In fact, the matter is much more complicated than that.

Here on Earth, we concern ourselves with more than extremes or range of atmospheric conditions. We are particularly interested in the changes from one set of conditions to another, and in the rate of change of these conditions. This is the essence of weather on Earth.

This change in atmospheric conditions, or weather, is important in planning the activities of our daily life, in providing for our comfort and in making operational decisions.

In our space program, for example, we monitor the weather very closely because space operations are influenced by and depend on weather for success. You will recall that John Glenn's orbital flight was postponed four times due to weather--once due to the weather over the launching pad and three times due to poor weather in an alternate recovery area.

Optimum launch vehicle operation is also dependent on weather, particularly on winds aloft. We have extensive upper air sounding programs involving both balloon and rocket-borne instrumentation--for providing data from which the characteristics of the atmosphere can be determined and the nature of its changes anticipated. Adequate weather observations are thus essential.

Weather Prediction

These weather observations form the basis for the next important element in the study of weather--that of prediction or forecasting. There are essentially four approaches used in weather forecasting (Figure 6):

1. The Analogue Approach: Daily weather maps are classified according to types. In any situation, an historical file of typed weather maps is scanned to find a weather situation identical or very similar to the current case. The forecast is then based on what happened in the historical case. There is an inherent assumption here that daily weather maps can be so uniquely typed.
2. The Statistical or Objective Approach: Here meaningful statistical correlations and probabilities are sought to relate a given set of weather parameters with the subsequent weather events.
3. The Physical or Empirical Approach: Physical reasoning is used to predict how the fronts, air masses, etc. will move and what the weather associated with them will be. This reasoning is usually based on past experience with weather in any location. One basic physical principle used is the tendency for weather events to persist more or less unchanged. This is called the persistence of weather and, in general, is a fairly reliable forecasting tool.
4. The Dynamical or Numerical Approach: In the dynamical approach, we use equations which describe the fundamental atmospheric processes and solve them by mathematical techniques. This provides us with a measure of the weather to come. In this process, it is assumed that the atmosphere is amenable to "deterministic prediction" and that given the necessary equations and input data, you can literally compute what will happen. Moreover, with the recent availability of very high speed, large memory capacity, electronic computing machines, we are reaching the point where we can produce a solution of these equations in a reasonable period of time.

Figure 7. lists the prediction equations to which I have just referred. Essentially, they are mathematical representations of: (1) the equations of motion--i.e., Newton's F=ma for the atmosphere, (2) the equation of continuity --or the law of conservation of mass, (3) the energy equation expressing the law of conservation of energy, and (4) the gas law relating pressure, density and temperature. These comprise a system of six equations in six unknowns--the three wind components, pressure, density and temperature.

In mathematical terms, these equations can be described as hyperbolic, non linear, partial differential equations. This means that, given the initial conditions, i.e., a distribution of the variables in space dimensions at a given time, and the boundary conditions, i.e., the effect of the underlying surface and extraterrestrial forces, these equations can be solved and the variables represented as a function of time. This means simply that one has forecast the state of the atmosphere into the future. With global data, it is estimated that this process can be iterated to produce meaningful forecasts of several weeks in advance.

Figure 8. lists the best estimate that we have today of the data grid distribution required for solving these prediction equations. A horizontal spacing of several hundred kilometers is necessary and a vertical distribution up to 9 levels, globally. The global requirement stems from the result that disturbances initiating in any part of the Earth can propogate over the entire globe in less than one week. Consequently, the atmosphere over the entire globe is involved in the long range prediction of the state of the atmosphere.

PREDICTION OF WEATHER ON MARS

By analogy with our experience on Earth, in order to plan for manned activity to and on Mars, we must become more fully familiar with the weather on Mars. A recent study undertaken by Robert Owen, of NASA's Marshall Space Flight Center*, concluded that "*...Knowledge of the gross features of the Martian surface appears to be fairly complete, but there is sharp disagreement on the atmospheric conditions... Formal design criteria for entry vehicles cannot yet be finalized because of the wide range of environmental parameter values.*"

As illustrated in the simple story with which I started this lecture, even a knowledge of the range of environmental parameters may not be enough. It might be critical that we know more specifically about the weather conditions on Mars at the place and at the time where and when manned activity is contemplated.

This involves being able to predict the weather on Mars. If we review the prediction approaches in use on Earth, we readily see that, of the four approaches described, probably only the dynamical approach will be applicable. The other three, the analogue, the statistical and the empirical are all based on a relatively long history of observations--something nonexistent for the Martian atmosphere. On the other hand, the dynamical approach is perfectly general and applies to any hydrodynamical system provided appropriate astronomical parameters are introduced in the prediction equations, i.e., radius of planet, its gravity, its atmospheric molecular weight, the solar input, etc.

As we discussed earlier, given an initial global distribution of the atmospheric variables, a prediction can be made of its future state. Thus our problem reduces to the acquisition of atmospheric data globally on Mars. In other words, we have returned to the problem of observations.

WEATHER OBSERVATIONS ON EARTH

Current Station Network

Again, we will proceed by analogy to the conditions for Earth. We first note that the problem of global observations is far from having been solved for Earth. This is due to the unsatisfactory distribution pattern of the observing stations. As you can guess, the distribution of the existing stations has been governed essentially by non-meteorological circumstances. Primary among these is the simple fact that almost all of them have been established where men live. The existing station network thus has come to cover inhabited land areas, islands and oceanic shipping lanes. This map (Figure 9) shows the world divided into 100 equal areas. The shading in each area represents the extent to which the density of observing stations is sufficient to provide the quantitative information required for dynamical prediction. Only in the darker areas of the illustration (less than 10%) is this density adequate. In most of the world, excepting only

*"*The Martian Environment,*" NASA TM-53167, November 19, 1964.

the United States, and the other indicated portions of the Northern Hemisphere, the density of observing stations is totally inadequate for the requirements of the dynamical or numerical approach to forecasting.

Under the current situation, then, we need a tenfold increase in density of stations over the world. To accomplish this by expansion of the existing measurement techniques alone is simply not feasible, both from an economic point of view and from the logistics point of view. A possible solution lies in the use of Earth orbiting satellites.

The Satellite As An Instrument Platform

The satellite as a system has two basic capabilities which permit its application to the solution of this problem. The first of these is its capability as a stabilized platform for carrying instruments for taking measurements of the atmosphere. For example, a polar orbiting satellite such as shown in Figure 10. provides for viewing the entire planet periodically as the planet rotates beneath it. Satellite-mounted sensors which view the Earth's atmosphere must view it only at a distance and therefore are disigned to measure the radiations in various portions of the electromagnetic spectrum.

Figure 11, shows what has been done in this area and what is being planned. TV cameras, operating in the visible portion of the spectrum, are able to provide global cloud cover distribution during daylight. This has been extremely useful for charting weather on Earth since clouds are, in essence, the signatures of atmospheric systems. We are thus able to identify, locate and track the course of atmospheric phenomena during the day. Nighttime cloud cover data are provided by IR radiation measurement devices. These provide, in addition, cloud top measurements, heat budget measurements, and temperatures of radiating surfaces, i.e., cloud tops or Earth's surface in absence of clouds. A flight in the near future of both an IR spectrometer and interferometer will provide the badly needed vertical temperature profile (above cloud tops) and measurements of the moisture and ozone concentrations. Finally, we anticipate that microwave techniques will yield temperature measurements below clouds and data on surface characteristics.

There is much missing from this list--most serious of which is the measurement of wind and its distribution with height. At present, we have no technique available for the measurement of wind directly by means of instrumentation on board a satellite.

The Satellite As A Data Collection And Relay Device

In order to be able to do this, we must consider the second basic capability of the satellite and that is to participate as a part of a global system for taking measurements by means of sensors immersed in the atmosphere. In this system, the basic meteorological parameters are measured directly by means of unmanned, automatic, instrumented platforms arranged in the atmosphere and on the Earth's surface (Figure 12). The types of platforms would include constant level balloons, automatic land stations, and fixed and free floating buoys. The satellite in a polar orbit views the entire Earth periodically every day. As it passes over the sensor platforms, it can make periodic interrogations of the platforms and record the measurements and location data.

As shown in the figure, the satellite would interrogate a platform and the time to respond to this interrogation would determine the distance of the platform from the satellite. This is represented on the chart in the inset where an arc of a circle is drawn at the computed distance from the subsatellite point. A second interrogation determines the location of the platform by intersection of the two arcs. In the case of a balloon, positions computed on consecutive orbits will determine the average wind speed during this period.

An experiment to test this system will be conducted on a future Nimbus flight, the satellite depicted in this figure.

POSSIBILITIES OF SYSTEMATIC EXPLORATION OF THE MARTIAN ATMOSPHERE

We have so far considered the following:

a. Weather comprises not only representative values of the state of the atmosphere and not only the range of these values (i.e., the extremes), but also the temporal changes from one state of the atmosphere to another.

b. Knowledge of the future weather is important to human activity in general, and to space activity in particular. This knowledge will be important in planning for the landing of man on Mars and his existence there.
c. Prediction of Martian weather can best be effected by the dynamical approach which has been described above. Required are data on the quantitative measurements of atmospheric parameters globally, both in horizontal extent and vertically.
d. These data are not yet available for Earth. Prediction of weather on Earth is aided by a large body of historical data not available for Mars.
e. Current space activities include programs for acquiring the necessary data for Earth. It is not expected that these techniques will become operational before the early 1970's.

Certainly, so far as the systematic observation of the Martian atmosphere is concerned, we cannot hope to do any better than our plans for Earth itself, and thus we might attempt to reproduce for Mars our accomplishments and plans for Earth. Accordingly, we might consider a satellite in polar orbit around Mars and a system of instrumented platforms distributed in the Martian atmosphere and on the Martian surface. This orbiter would carry instruments such as cameras, spectrometers and interferometers, and a subsystem to interrogate the instrumented Martian platforms, locate them and record the data on board for transmission to Earth.

The unmanned orbiter system design could be based on the Nimbus experience around the Earth and could have similar control and stabilization systems and sensors. The stabilization problem could be simplified with a manned orbiter, however, in this case, we would sacrifice duration in orbit.

The Martian atmospheric instrumented platform would present a serious logistics problem in terms of maintaining a suitable distribution of platforms. Surface instrumentation would have to be soft landed in order to avoid impact destruction. The atmospheric balloons could be released on descent of a capsule, or released from the capsule after landing. In the Mars case, we probably will not be too concerned with the hazard to aviation which these balloons represent on Earth. From that point of view, it would be more desirable to use the balloon platforms in studying the Martian weather than on Earth.

There would be a serious data transmission problem in view of the tremendous volume of data to be expected. In all probability, some on board analysis techniques will have to be devised in order to reduce the amount of data transmitted back to Earth from the orbiter. Alternately, in the manned orbiter case, the scientist aboard the orbiter might perform the analysis.

CONCLUSIONS

At this time, we are not sure of the specific weather elements that will affect man's landing on Mars and his existence there. It would appear, from available temperature measurements, that we shall be able to cope with the temperature range to be expected. Moreover, current estimates of low atmospheric surface pressure and of insignificant water vapor on Mars would seem to eliminate most of Earth's familiar weather problems excepting that of surface winds. These winds could affect man's landing and existence on Mars, either as a dynamic force in themselves, or as producers of low visibility as dust carriers. Thus, it may turn out that the prediction of winds on Mars is really the only critical weather related activity that will concern us.

At any rate, weather on Mars must be one of the important elements to be considered in the manned Mars landing program and its study should precede that landing. A useful place to begin is by considering the observational and prediction techniques being developed for Earth and extending these techniques to the Mars problem.

I would like to conclude with some remarks on weather control. One cannot overstate the potential importance to mankind that effective weather control could provide. Our efforts towards weather control here on Earth are complicated by many factors. First, our incomplete knowledge of the atmosphere and its behavior prevents us from conducting meaningful scientific experiments in this field. It is difficult to assess the results of any experiment because we do not know what the atmosphere would have been like had we not performed the experiment. Moreover, we are cautious of introducing copious energy sources, e.g., nuclear

energy, into the atmosphere because of the potential danger to mankind from unforeseen results. Additionally, there is a social problem introduced by attempting to alter weather in any one locality since it would be difficult to get agreement on what weather would be best for all. For example, the farmer's desire for rain would conflict with that of the out of doors sports event promoter.

As our knowledge of the Martian atmospheric behavior improves and our ability to predict its weather becomes more proficient, we could conduct a range of useful weather control experiments on Mars -- unless, of course, we find Martians there who would seriously object.

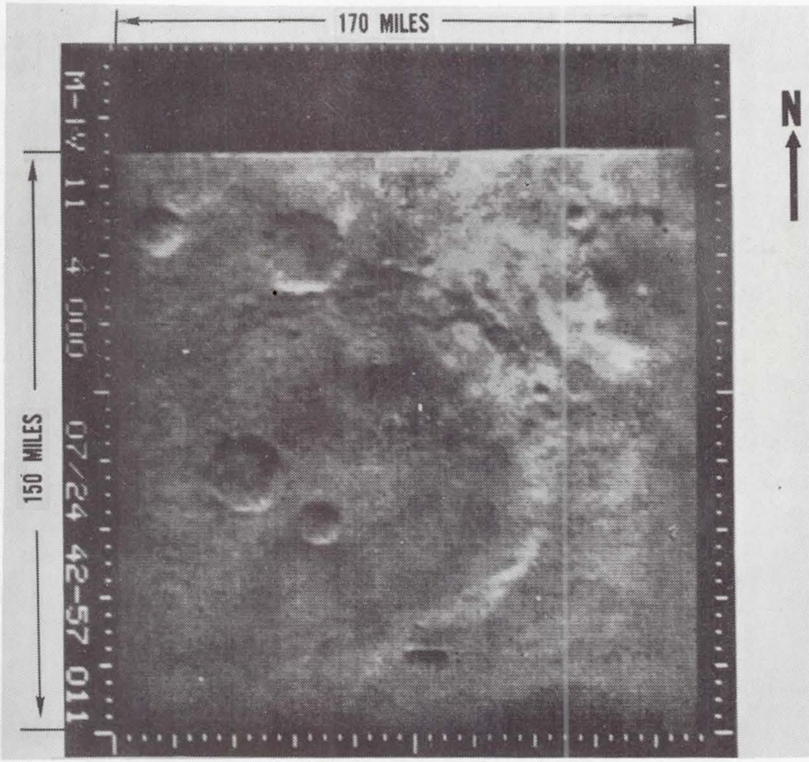

FIGURE 1

Mariner IV Picture No. 11 Atlantis, Between
Mare Sirenum And Mare Cimmerium

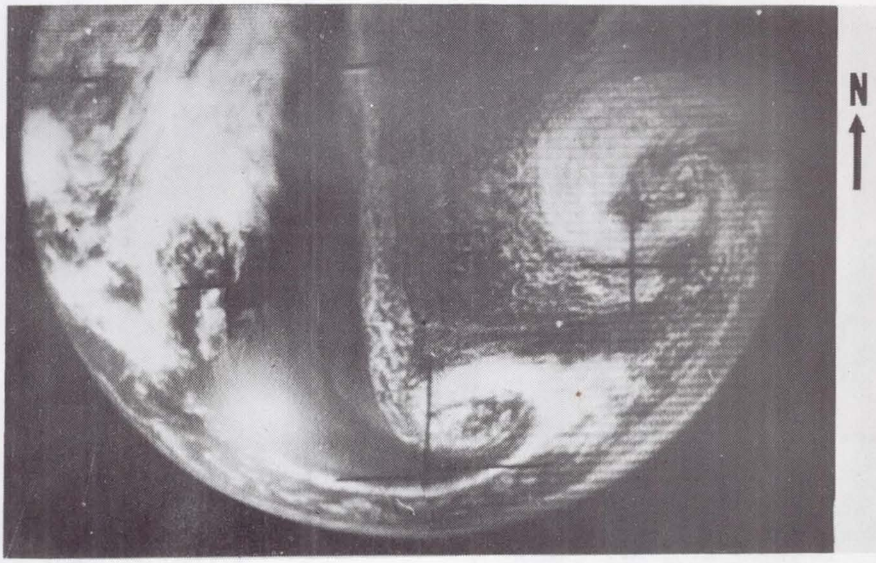

FIGURE 2

Planet Earth From 2000 km

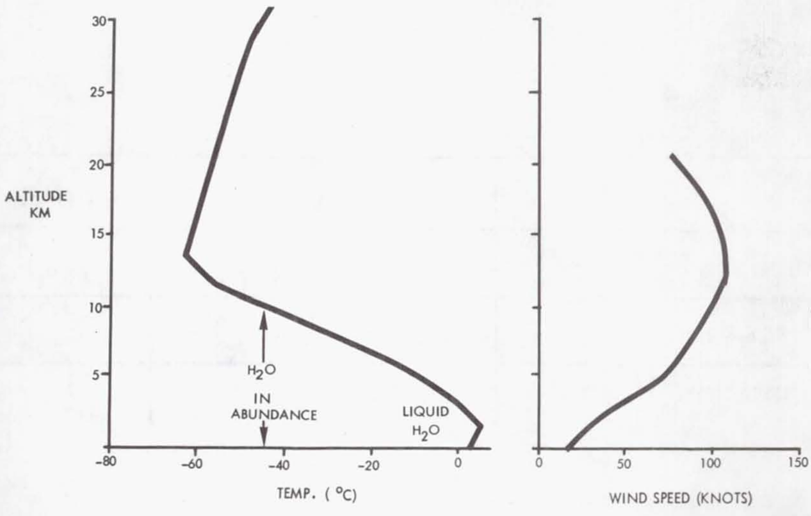

FIGURE 3

(WAP) Temperature And Wind Vertical Profiles

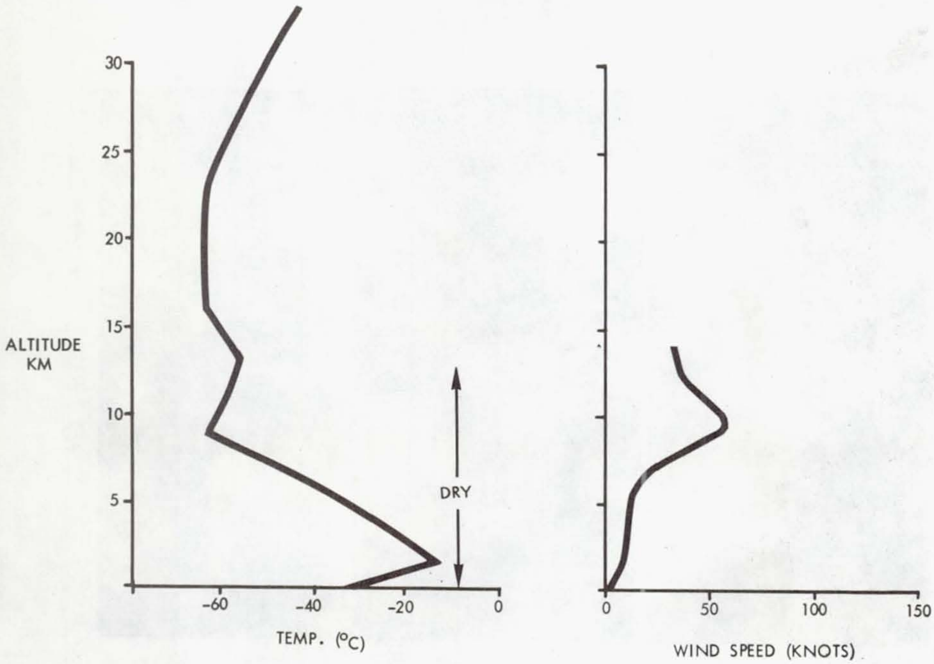

FIGURE 4

(DAP) Temperature And Wind Vertical Profiles

	WAP	WAMEL	DAP	DAMEL
TEMPERATURE	5 °C	20 °C	- 30°C	45 °C
SURFACE WIND	20 Kn	125 Kn	0	0
PRECIPITATION	.05"/HR	> 5"/HR	0	0

FIGURE 5

Comparison Of Probe And Lander Measurements Of Surface Weather

BASIC WEATHER PREDICTION TECHNIQUES
(EARTH)

1. ANALOGUE
2. STATISTICAL
3. EMPIRICAL
4. DYNAMICAL

FIGURE 6

DYNAMICAL PREDICTION EQUATIONS

3 EQUATIONS OF MOTION: $\rho, \vec{V}, p = f_{1-3}(x,y,z,t) + \vec{F}$

EQUATION OF CONTINUITY: $\rho, \vec{V} = f_4(x,y,z,t)$

ENERGY EQUATION : $\vec{V}, T, p, \rho, = f_5(x, y, z, t)$

EQUATION OF STATE : $f_6(\rho, p, T) = 0.$

$$\boxed{\text{SOLUTION: } \rho, \vec{V}, p, T = g(t)}$$

FIGURE 7

INITIAL CONDITIONS

PARAMETERS - WIND, TEMPERATURE, AND
MOISTURE, AND THEIR
VARIATION WITH HEIGHT (OR PRESSURE)

ON A 500 TO 800 Km WORLDWIDE GRID

FOR 2 TO 9 ALTITUDE LEVELS

FIGURE 8

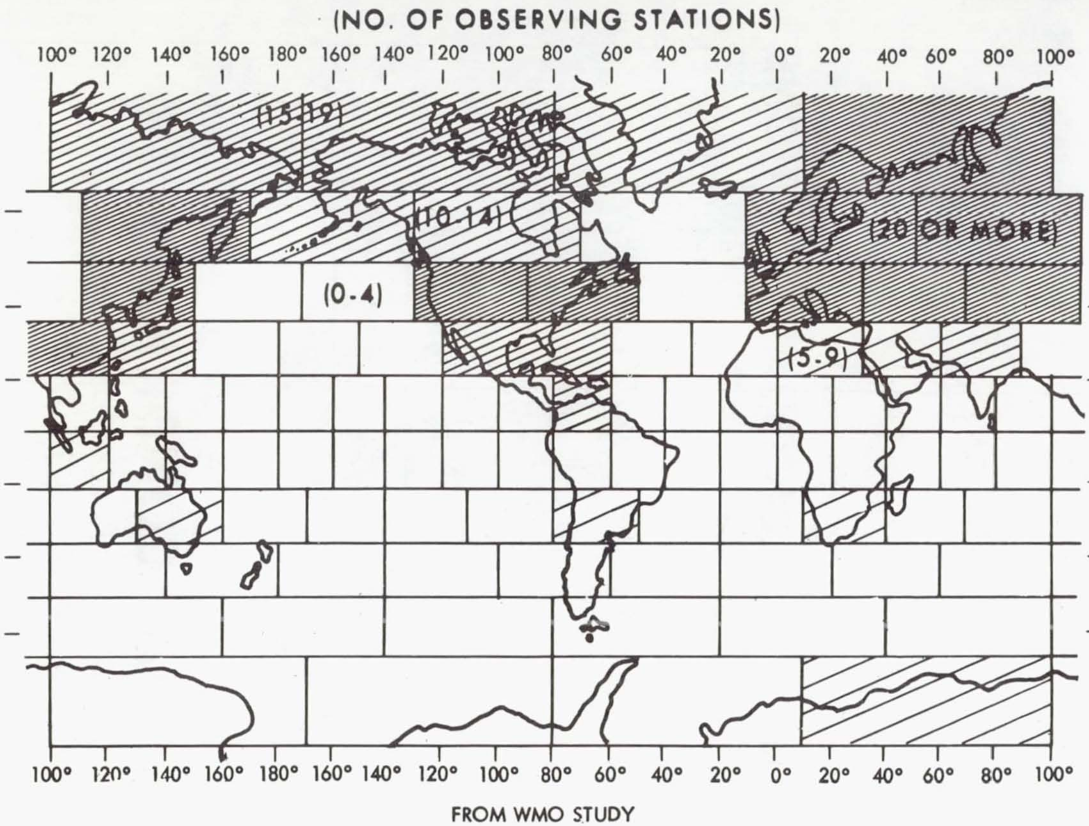

FIGURE 9

Areas Of The World Where Upper
Air Observations Are Adequate

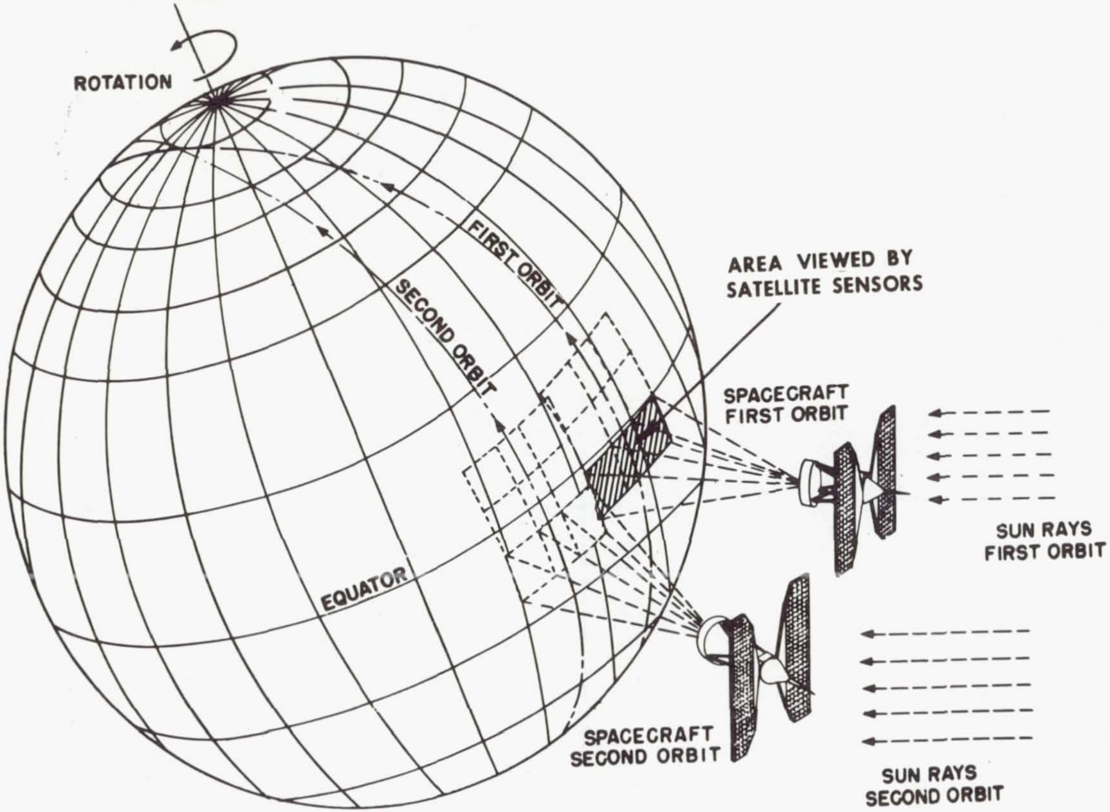

FIGURE 10

Polar Orbiting Satellite Viewing
An Entire Planet Periodically

THESE BANDS OF THE SPECTRUM	SENSED BY THESE DEVICES	CAN PROVIDE THESE KINDS OF INFORMATION
1. VISIBLE	CAMERAS	CLOUD COVER DISTRIBUTION - DAY (ALBEDO)
2. INFRARED	IR RADIOMETERS	CLOUD COVER DISTRIBUTION - NIGHT CLOUD TOP HEIGHT MEASUREMENTS HEAT BUDGET CLOUD TOP TEMP. MEASUREMENTS SURFACE TEMP. MEASUREMENTS - CLEAR AREAS
	IR SPECTROMETERS IR INTERFEROMETERS	VERTICAL TEMP. PROFILE (ABOVE CLOUDS) MOISTURE CONTENT, OZONE CONTENT VERTICAL TEMP. PROFILE (ABOVE CLOUDS)
3. MICROWAVE	SPECTROMETERS	SURFACE CHARACTERISTICS VERTICAL TEMPERATURE PROFILE

NASA SF65-15894
8-16-65

FIGURE 11

Electromagnetic Radiations
As Viewed From A Satellite

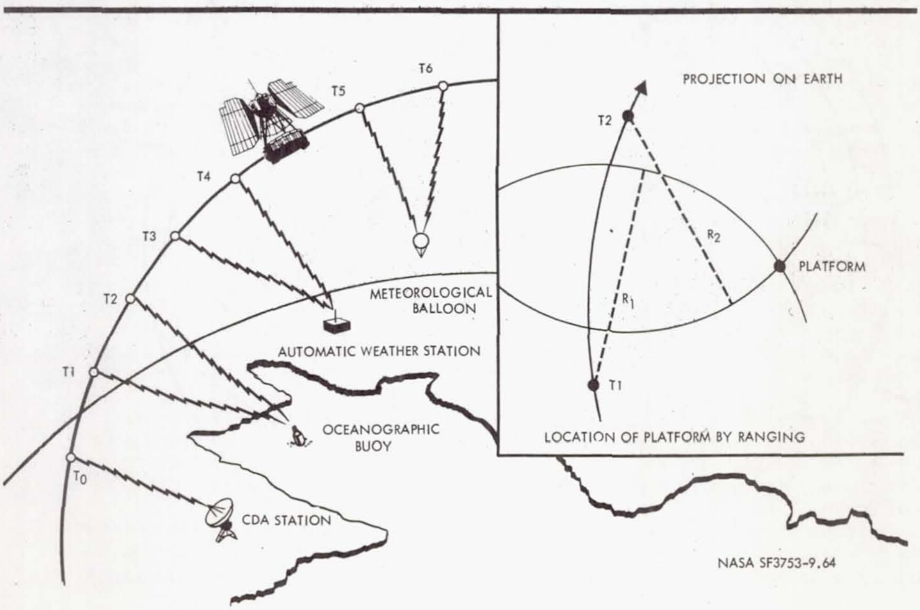

FIGURE 12

Interrogation, Recording
& Location Subsystem

THE POSSIBILITIES OF LIFE ON MARS

By

Frank B. Salisbury

Colorado State University

At this stage of our exploration of Mars, we have learned a great many facts, all of which have a great many implications. But we still can say nothing conclusive about the possibilities of life on Mars. Indeed, we can discuss intelligently each of some four different sets of characteristics relating to the face of Mars. We can even make fairly good cases for the two extremes of this continuum of hypothesis: that Mars is a completely dead world, or that Mars supports a complete biota, including an intelligence capable of highly advanced technology!

A dead, lunar-like Mars. We must begin our discussion of life with life as we know it, for it is the only kind of life we know. But we must also realize clearly that life on Mars could easily be different in many ways from life on earth. But in how many different ways? Could life exist in areas where the temperature never gets above the freezing point of water? (Ground temperatures above the freezing point of water have never been measured at certain points on the planet which are within the markings discussed below). Could life exist in an atmosphere with a pressure of 10 to 20 millibars at the surface, as postulated by Kuiper and substantiated by the Mariner IV occultation experiment? Could one have a balanced, cycling ecology in the absence of oxygen? How could life develop where water is so limiting as to be almost nonexistant? Wouldn't the ultraviolet light or shorter-wavelength ionizing radiation be lethal to life? How could life ever come into existence on a planet which has always been dry and cold, as seems to be implied by the Mariner IV photos, showing a crater-pocket, non-eroded surface?*

Considering all these things, it would seem that Mars is a dead planet and always has been, a cold dry desert, whipped by winds of high velocities, blowing in an almost non-existant atmosphere. In the light of our modern science of biology, this seems to be the picture of Mars which best fits the facts. In the minds of the majority of scientists living today, this is probably the most reasonable and compelling picture. They may venture to other ideas to be discussed below, but they are quite likely to return to this one.

But we have not discussed all the facts. There are the Martian markings, large areas of the planet which intensify in color as the polar cap receeds towards the pole in the spring (see Figs. 1 and 2). Furthermore, this color change occurs first in proximity to the polar cap, moving then towards the equator with the coming summer, as though it were dependent upon water being made available from the melting polar cap (which probably does consist of water and not frozen carbon dioxide, as was once thought). How can all this occur on the dead planet described above?

Various theories have been put forth to try to explain how. Yet none of them has really been very satisfying. According to measurements and calculations, the increase in relative humidity with the coming of summer must be

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*We must note that the "origin-of-life argument" is far from conclusive. To begin with, we are far from certain as to how life originated on earth. Furthermore, life coula have been transplanted to Mars in varicus ways, such as in earch crustal material blasted to escape velosity by the impact of a large meteorite on the earth.

extremely slight. The humidity which has been detected was observed only above the polar cap itself. What chemical could change color with such a slight increase in relative humidity?

As a matter of fact, how can increasing humidity account for the changes in any way? Would not the humidity be rapidly spread around the planet by the winds of 30 to 125 miles/hour often observed blowing the dust? Why should the color change move only at a rate of some 16 miles per day, very evenly and regularly from the polar cap to and slightly over in some cases, the equator? It is as though liquid water were being moved from polar cap to equator. But then water doesn't flow on a level surface of soil, and present indications are that the dynamic elevation* at the equator may be as much as 16 km higher than at the poles - an elevation difference which would be of profound significance in considerations of temperature and atmospheric pressure, as well as the flow of water or atmosphere.

Are the color changes due to blowing volcanic dust as was once postulated? We don't consider this idea much anymore, since it fails completely to account for the direction and the rate of change.

Are the color changes due to the interesting physics and chemistry of the oxides of nitrogen? The polar cap might consist of the brilliant white nitrogen tetroxide, which in spring sublimes and changes to a brownish nitrogen dioxide, flowing in depressed areas of the planet towards the equator. But the Mariner photos do not show the markings to be depressed areas, and careful spectroscopic observation fails to detect any oxides of nitrogen in the Martian atmosphere, a result which is conclusively fatal to the theory.

So it is the markings which cast the most doubt on the first face of Mars, the dead and lunar-like planet. Yet to modern, conservative science, this is still an appealing picture. And we do have the models of the oxides of nitrogen or the hygroscopic chemicals which change color. These theories fail to account for the markings on Mars, but what similar theory have we so far failed to conjure up? Will we be surprised at our own stupidity when we some day learn the facts about the color changes? Why, for example, in all our postulating, didn't we postulate that the surface of Mars would be covered with meteorite impact craters? In retrospect, many observations fit this picture (e.g. the "oases", Slipher's "spots" on the maria), and the suggestion had been made,* but it was essentially ignored by the scientific community at large.

Primitive plants on Mars. Although many scientists may feel that the above picture of Mars has the greatest likelihood of being the correct one, they are willing to consider that the problems related to the markings may be solved by the presence of some sort of life on Mars. Some living, growing think might account for the dynamics of the Martian markings, including their ability to re-emerge in a matter of a few days after being covered by the blowing, yellow dust.

But in the spirit of our scientific conservatism, a spirit which has paid great dividends in dispelling superstitions and thus advancing knowledge, the scientist who is willing to consider life on Mars will almost surely qualify his ideas by saying that it must be a primitive form of life: bacteria, algae, mosses, the tough and resistant lichens. The implication is that "primitive" life forms are better able to survive in extreme environments, although they would be quite restricted in their growth manifestations. This is, in fact, the official position of a special committee of the National Academy of Sciences.

Yet everything about the markings seems to speak against a struggling, barely surviving, limited form of life, an earthly thallophyte, transplanted to Mars and growing there as we intuitively think it should grow there. The observations of the markings seem to imply a flourishing, well adapted life form.

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*Mars would be expected to bulge at the equator in response to the centrigugal forces caused by its rotation, but the observed bulge seems to be 16 km more than the rotation could produce.

*According to Richard Sloan of JPL, the astronomer Clyde Tombaugh described the craters in a letter to the JPL scientists received just before the Mariner photographs were taken. During the VP conference a booklet fell into my hands in which craters on Mars are discussed in detail and with prophetic accuracy by a writer of popular science in 1944: Donald Lee Cyr, 1944, Life on Mars, Desert Magazine Press, El Centro, California.

Consider the density of the markings. When they are in full color development and when seeing is good, they are not difficult to see. Indeed, when photographed through a red or orange filter, they stand out in sharp contrast to the surrounding areas. It is as though whatever they consist of covered the ground rather completely. Some of the background reddish-yellow color does show through the markings, indicating that the cover is not absolute (as it would be, if it consisted of nitrogen dioxide or something similar), but it is very nearly so. In our deserts, where both primitive and advanced plants really struggle for survival, they often cover the ground so sparsely that they can not be seen at all from elevations which are great enough that they can not be visually resolved as individual plants (see Figs. 3 and 4). That is, they are so few and far between that their contribution to the general background color is negligible. This is not so on Mars.

Most of the primitive plants envisioned on Mars do not even show the color changes typical of the markings. The lichens, for example, show virtually no color change with season. Perhaps 2 unicellular algae could nearly die out in the fall, then grow and multiply, progressively covering the ground with the approach of spring and summer, but we have no earthly precedent for such an activity where conditions are so cold and dry.

Speaking most strongly against a barely surviving form of life on Mars are the observations of the invasion of desert areas by markings. This has occurred off and on during the past century of careful Mars observation, but a most spectacular example occurred in 1954 when a marking the size of France (the Nodus Lacountis) appeared in a desert region where nothing could be seen in 1948. The beginnings of this region were discernable on plates taken in 1952. As a matter of fact, a darkening had been seen for a few days at that location in 1926. If this was life moving into the desert, it was not some form almost on the verge of extinction!

So the compromise position suffers from being just that. It may suit the mind of the scientist. He says: "Life can't exist under the conditions on Mars, but the markings seem to indicate that life is there anyway, so I'll compromise by assuming that there is just a little bit of very primitive life." But this position does not fit the observational facts very well. If the markings on Mars really do represent life, then it would seem to be a well adapted life, one which is thriving and flourishing.

Isn't this what our science would <u>really</u> predict? Don't the adaptive processes through time result in a progressively more intricate and complete suitability of life for its environment? Isn't this the real lesson of our modern biology? I can imagine that a form of life matched to extreme cold and dryness might not survive on our planet under the predation of other organisms able to live nearby in more moderate conditions. But on Mars the severe conditions are everywhere. Wouldn't life, if it exists there, become well adapted to them?

<u>Thriving, advanced plants on Mars</u>. It is exciting to contemplate eventual discovery and study of a flourishing, well adapted, truly Martian form of life. We should now do all that we can to find out about the limits of survival for life on earth.

Pioneer experiments of this type are presently being carried out in several laboratories. For example, consider some of the unpublished results of Dr. Sanford Siegle of Union Carbide and Carbon in New York, summarized for me a week before the VPI Conference:

(1) A mealworm (Tenebric molitor) in the larval stage has survived for 10 weeks (10% survival of the initial population) in an atmosphere (1 atm pressure) of 5% CO_2, 95% N (less than 0.01% O_2), and a dewpoint of -60°C, with day temperatures of 20°C (12 hrs) and night temperatures of -25°C. They are placed on a medium of dry farina meal. The animals die if introduced into these conditions in the puppal or adult stage or during the warm phase of the cycle. They are completely frozen during the cold part of the cycle, and this is fatal if normal oxygen levels are present. Lack of oxygen is fatal if the organisms are not frozen each 24 hours!

(2) Several seeds have been allowed to imbibe water in the absence of oxygen (O_2L0.01%) and then germinate in dry, oxygen-free conditions at reduced pressures. Examples of minimum pressures for germination are:

Various species (e.g. morning glory)	100 millibars
<u>Dianthus barbatus</u> (a wild carnation)	30 millibars
<u>Celosia argentea</u>	25 millibars
Cucumber	16 millibars
Winter rye	16 millibars

Most species were germinated at 20°C, but winter rye was germinated under the temperature and other conditions described under (1). Typically, seedings would grow for two or three weeks and then be killed by flourishing molds, most of them normally saprophytica and not parasitic.

(3) Under reduced pressure (16 millibars), anoxia (O_2 less than 0.01%), extreme temperature (20°C day, - 25°C night), and low humidity (dew point at -60°C) several fungi have been observed to grow and flourish, completing their life cycle on seeds and seedlings used in (2). These include Penicillium sp., Botrytis sp., Aspergillus sp. and A. niger, Mucor, and Torula sp. Many bacteria, especially Pseudomonas sp., flourish under simulated Mars conditions, as well as in atmospheres of ammonium, in saturated salt solutions, etc.

(4) A dwarf palm (Oenanthe) survives more than 19 days before bleaching at 12 hrs 20° and 12 hrs - 20°C, providing oxygen is absent. At 21% O_2, death occurs on the first day. The Martians, then, might consider our high oxygen levels as poisonous and quite deadly!

(5) Common soil bacteria can be grown in saturated lithium chloride. This implies that those organisms could extract water from air with a vapor pressure of less than 0.1 mm at $T = 265$ to 270°K.

Miss Judith Herr, at St. Petersburg, Florida has also tried to grow a number of species under simulated Martian conditions (she finally duplicated virtually all the postulated Martian conditions for a period of time equivalent to an entire Martian year). Again most species died, yet a moss (Funaria) not only survived, but became so well adapted that it produced spores and lived through several generations. As a matter of fact, during about three years of increasingly precise Martian simulation before the simulated Martian season, Funaria gradually became changed in appearance, and when removed from the Martian simulator, it would promptly die!

Dr. Morris Cline, working in my laboratory, has surveyed a number of species for their resistance to ultraviolet light. Again, most are quite sensitive, but Austrian pine seedlings grew 635 hours under the intensities of ultraviolet encountered on Mars, showing only slight needle damage for the experience. Other, almost equally resistant species were also found. The leaves of the most sensitive species we can find die after three to four hours of exposure.

Now that we take the time to look, we find that life under natural conditions on earth is far better able to stand extreme conditions than most of us might have previously realized. A red-colored green algae, for example, grows and flourishes in mountain snow banks at 0.0°C and below. A buttercup (Ranunculus adoniis) pushes its flowers up through these same snow banks. Blue-green algae may grow in hot springs near the boiling point. Bacteria are known which can metabolize sulfur or iron, and certain types are known which can grow in gasoline tanks (usually in assoclation with drops of water) or even phenol! Living things grow near the bottom of the ocean where conditions are what we might be tempted to call extra-terrestrial: cold, total darkness, pressures of tons per square inch. In certain respects, some of these conditions are worse than the worst of Mars.

For some time now, many astronomers and exobiologists have considered the possibilities that life might find suitable conditions in certain micro-habitats on Mars. Caves, surface fissures, hot springs, etc. have been suggested as locations where moisture might collect and temperatures might be somewhat higher. Such conditions would hardly account for the markings on Mars, but with the recent revelations by the Mariner, we can now visualize the bottoms of craters as potentially suitable micro-habitats for life -- and on a scale that might begin to account for the markings. Indeed, most of the pictures showing many craters are of areas within the markings (see Figs 5 to 8; picture No. 7 is an exception, being of a desert area but showing many craters). In craters with depths of over ten thousand feet, atmospheric pressures at the bottom would be approximately twice as high as near the rim, wind velocities would not be nearly so high as at the surface, and temperatures would be more moderate. In deep, small craters, the sides would radiate some heat at night to the bottoms, so that temperatures would remain higher during the night. Frost and snow, collecting on the crater rims (as indicated in some of the Mariner photographs - see Nos. 11, 13 and 14) might provide liquid water which could run down to the bottoms. Mariner photograph No. 11 may indicate the erosional pattern which would be expected on this bases (see Fig. 10). A drainage pattern seems evident from a high, possibly snow-covered plateau near

the rim of a very large crater. Indeed, in the porous material caused by the meteorite impact, ground water might collect from outside the rims and be available to deep rooted vegetation.

It is possible to think of several ways in which life on Mars might become adapted to the extremes of temperature, atmosphere, and drought, and protection from ultraviolet light is apparently not difficult, as indicated by our experiments mentioned above. Certain alpine plants are known which can freeze at night, thaw out the next day, and continue to grow unharmed. This would seem to be the simplest mechanism of protection against the very low temperatures, but protoplasmic solutions with much lower freezing points than earthly protoplasm, leaves which roll into tight, insulated tubes or balls, and other mechanisms might also be imagined. I have suggested that Martian organisms might change the oxidation level of oxygen in the compounds of the Martian soil by a photosynthetic process of some sort, never releasing it as a gas, but making it available as an energy source for certain reactions. On the other hand, some other material might perform the primary oxidation-reduction of Martian metabolism. Nitrogen, thought to be present in the Martian atmosphere, might even perform this function, although it would be somewhat less efficient than oxygen.

I have also suggested that water might not be the primary solvent of Martian protoplasm, this being replaced by some metabolicly synthesized liquid (admitadly a rather unlikely substance). Water might act more as a "vitamin", vitalizing Martian life as its concentration in the Martian atmosphere increases by a few molecules per unit volume. But as indicated above, this would not account for the slow and regular progression of color change from polar cap to equator. Temperatures would be very suitable for life near the equator. Wouldn't moisture in the vapor state move there rather rapidly? The color change reminds one of liquid water being moved gradually on or below the surface.

How could liquid water move from the polar cap to the equator on the surface of Mars? In terms of natural mechanisms, this seems completely unfeasible. If anything, the elevation gradient from pole to equator may be uphill. Although we think about it reluctantly, we might consider a technologically constructed pumping and distribution system (possibly underground). Is there anything about the markings which might imply an agricultural system, operated by an advanced form of Martian intelligence?

Intelligence on Mars. The area in the desert the size of France which appeared in 1954 could be a reclamation project, but it could also rather easily be something more "natural".

Although exact location of the photographs on the Martian disk may modify the idea, one rather unexpected result of the Mariner photographs may prove very significant in our speculations about life on Mars and may apply to the problem of intelligence although the presence of craters is roughly correlated with the markings on Mars, these markings generally do not correspond to the topographical features revealed in the photographs. Number 7 shows many craters where no astronomer has ever indicated a marking, and Numbers 8, 10, and 11 each show areas indicated by astronomers as the edges of markings (compare Figs. 6 to 11 with Fig. 5). Yet no features on the photographs indicate these edges. There are no fault lines between markings and surrounding "deserts". Indeed, in some cases and providing that the maps and the photograph locations are correct, the edge of the marking seems to cut across crater systems, and this for craters so large that their general shape should be visible in the telescope.

How could Martian vegetation fail to be correlated with topography? On earth certain types of vegetation (e.g. forests, grasslands, tundra, etc.) clearly are restricted to certain elevations at a given latitude. On the Mariner photographs, the bottoms of certain craters (Numbers 7, 8, 10, 11? and 13; see Figs. 8, 9, and 10) do appear darker than the surroundings, although they are not in shadow (this may be vegetation as suggested by the micro-environment hypothesis mentioned above), but aside from this, markings and topography bear no obvious relationship to each other. We have only one precedent for such a situation on earth: a managed vegetation! If the markings on Mars are a managed agriculture, then nearness to "cities", underground water lines and other factors might be more important to the distribution of "farms" than topography!

Probably the most significant feature of Mars which might indicate intelligence is the network of fine lines called "canals". These lines criss-cross the deserts and the maria as well, usually forming intersections at spots called "oases" (which we now can expect to be craters). Since Schaparelli reported seeing this network in 1877, controversy has keynoted discussion of the topic. The Mariner photographs show no clear indication of certain canals which should appear in them, although in one or two cases certain features seem to be present in the proper places, and in at least one other case, a linear feature extends over two overlapping pictures (Nos. 11 and 12; see Figs. 10 and 11)

We are faced with three possibilities: first, the canal network or anything like it simply fails to exist; second, the lines are optical illusions brought about in the eye of the observed as certain small details converge or as lines appear to connect up random and separate points (Frence astronomers hold to this view); or third, the canal network really does exist as a series of connected lines on the surface of the planet. In the Mariner photographs these might have been invisible because they also show the seasonal color change, and most are in the southern hemisphere which was experiencing winter at the time of the Mariner fly-by. (But some of the pictures were in the northern hemispheres, and three of them might have shown canals, but they did not show them clearly.)

I can't quite believe the network is nonexistant. Several years ago some new data seemed to indicate that the cell membrane was a nonfunctional entity, and most physiologists accepted this new conclusion without question. One biologist however, pointed out that the evidence for the functional features of the cell membrane had been accumulating for over a century, and that we should be a bit careful in letting one new line of reasoning negate all the previous evidence. Don't we have the same situation with the canals? For a hundred years certain astronomers have been seeing and mapping this network. Do we discard it because it doesn't appear on the Mariner photographs?

Surely the network has some basis in fact. Whether it consists of lines or disconnected points remains a mystery, but something is there which one day will be explained. What does it mean? Assuming that the network as drawn by astronomers such as E. C. Slipher is essentially valid (see Fig. 12), what are the implications of the Mariner photographs?

To begin with, it is impressive that the distribution of craters does not match the canal network in any clear way. If the location of the photographs on the map is really accurate, then the craters clearly do <u>not</u> constitute the separate points postulated by the French to account for the canal network. In a few cases canals have been drawn in such a way that they clearly cut right across certain craters (while other craters in the near vicinity bear no relation to the canal). So all the "natural" explanations for the canals postulated so far must fall, because they depend upon topography. The craters do not explain them, and there are no huge earthquake cracks, aroded canyons, fault-block valleys, rays of exploded material (as on the moon), or other topographic feature which might correspond to the position of the canals.

A sufficiently advanced technological society, on the other hand, might run its underground pipelines and build its roads along straight-line, great-circle arcs of the planet's surface, the shortest distance between centers of population - and in spite of intervening topography such as craters! This is the kind of fantastic, science-fiction explanation which distrubs most of us--but which fits the facts and seems to solve the anomoly of the Mariner photographs as contrasted to the astronomer's maps.

If we once allow ourselves heretically to consider an advanced technological society on Mars, then we soon begin to wonder about some other strange observations relating to the planet. This proves to be interesting and lots of fun, so long as we do it in the spirit of uninhabited speculation and without taking it too seriously. For example, brilliant flashes of light, called flares, have been seen on the surface of Mars on various occasions for many decades now. They are as brilliant (sixth magnitude from earth) as they would be if they were produced by a high-megaton hydrogen explosion, they last about as long (from several seconds up to about five minutes), and they are sometimes followed by a rapidly expanding white cloud. Surely they are <u>not</u> hydrogen bomb explosions (the coincidence in historical time would be too fantastic), but what are they? Volcanoes do not seem to act this way, nor do the Mariner photographs show volcano-type craters (resolution is such that this is not conclusive). Are they meteorite impacts, the kind of event which probably caused the craters? This seems possible, but the duration of the flash is too

long. Besides, the chances of this being seen are very small. One impact every 40,000 years would produce 100,000 craters in four billion years.

The two satellites of Mars (see Fig. 13) have many of the features of artificial satellites and virtually none of the features of the natural satellites in the solar system such as our moon, and the moons of Saturn and Jupiter. They are small, about 5 and 10 miles in diameter if their albedo equals that of the moon; much smaller if their albedo is higher. They are close to the planet, moving in circular, equatorial orbits. The inner one may even be responding by friction to the extremely thin atmosphere at its elevation of nearly 4,000 miles. To do so it would have to have an immense surface and a very small mass. A thin hollow sphere, such as our Echo satellites, would have these characteristics

All this leads to the argument which is often considered to be most conclusive against intelligent life on Mars: such an advanced technology would surely be capable of space travel (certainly the satellites would imply this, and the flares might imply that they are there now, not that they advanced to a high level of achievement and then died out millions of years ago, as has been suggested). Why haven't they visited us? The implied idea that they have not is often presented as the final and conclusive argument against intelligence on Mars.

But is it so final and conclusive? Or is there a chance that they have visited us and observed us, but for reasons known to them alone, they prefer to make no direct contact with us? I do not know the answer, but the question led me to an involved study of the unidentified flying objects. Space will not allow the proper documentation, but my present conclusions are as follows: (1) Many sightings of UFOs are sightings of natural or man-made objects which are misinterpreted by the observer (I have seen two "flying saucers", one proved to be the planet Venus, and the other a huge plastic weather kite). (2) Others are hoaxes or clear misrepresentations (the so called "contactees", individuals who claim communication with the saucer people, probably fall in this category). (3) Some sightings may be psychological phenomena (in the majority of cases, this seems very unlikely), and (4) It is even possible that secret weapons or vehicles under development have been classed as UFOs (but this will not account for some very excellent sightings reported towards the end of the last century or even earlier). So the noise level is high. That is, there are many reports of objects in our skies which are not familiar to the observer but which are clearly not space ships from Mars or some other extraterrestrial location.

Yet many other sightings, (literally thousands - over the entire world) defy classification into these comforting categories. Too much detail is observed (a hemispherical object standing in a meadow with a door and windows, leaving tracks in the ground after it departs at high speed, for instance), and too many witnesses are present (certain objects clearly not natural nor man-made have been seen on more than a dozen occasions by hundred to thousands of witnesses). Photographs have been taken in the presence of many witnesses who back up the story (Fig. 14), and objects have been seen visually from the ground and the air and at the same time followed on radar. Cars have been stalled by a UFO; holes have been left in the ground and magnetism in fences; and vegetation has been crushed or burned.

Certain correlations have been noticed. For eight years the number of sightings was inversely related to the distance to Mars (Fig. 15). The pattern of sightings shows consistancy (first UFOs followed topographic features, then military installations were visited, then reservoirs, etc). Sightings sometimes occur on a single day along great circles ares of the earth's surface (Fig. 16). Other evidences have also accumulated.

I am completely unsatisfied by many explanations put forth by the United States Air Force and by air forces in other countries following the reports of such objects. Often these "explanations" are as thin as tissue, yet once they have been announced, then the case is dismissed and written off as solved. A trained scientist should, think for himself, rebeling against this dictation by authority. Study carefully all aspects of the reported sighting, and see if you can then always dismiss it as easily as the Air Force officials would suggest.

SUMMARY

So what can we say about the possibilities of life on Mars? As the arguments have developed above, each fact has seemed to lead more and more to the

conclusion that Mars supports not only life but an advanced technology! Yet we are still faced with the bleak facts of the Martian environment. To say that Mars supports a flourishing biota, even an intelligent population of advanced beings, is to say that much of our current theory about life and how it developed and the environmental conditions under which it can grow and flourish is false or at least incomplete. We know that life on earth is not life on Mars, but then basic physics and chemistry, the backbone of biological function, should be the same anywhere. So there is much justification for the attitude of today's scientist. According to what he has been taught, life just couldn't amount to much on Mars.

But those pesky markings remain. They demand explanation and understanding. They look and they act like life. How could they be anything else? So we begin to discard our preconceived ideas and remove all limitations on our speculations. If we will do this honestly and with imagination, we seem to be driven to the conclusion that the markings of Mars represent an advanced and flourishing form of vegetation. If we allow ourselves to go this far, then before we know it we are talking about canals and flares and satellites--- and even UFOs! Is this justified? Who can say? We can only wait and see. Is Mars a dead and moon-like planet, whipped by the winds of a thin atmosphere? Or has some form of life managed to conquer these conditions, developing mechanisms which allow it to flourish, even to think and develop technology? We can only strive with all the ingenuity of our own technology to devise ways and means for finding out. The exploration of Mars is one of the most challenging endeavors ever faced by man.

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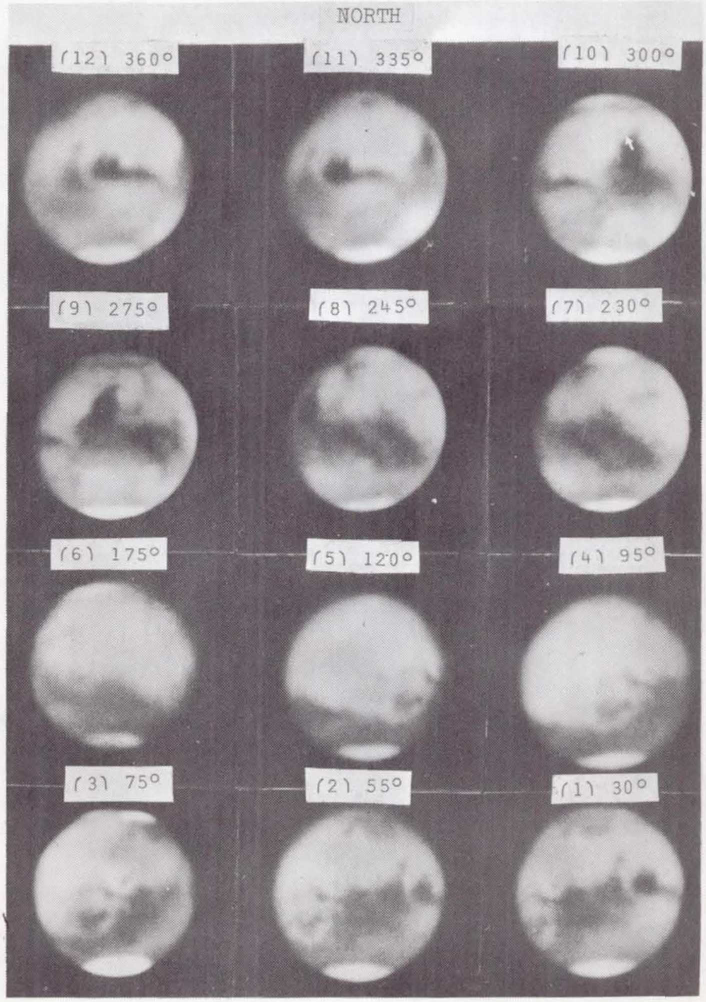

FIGURE 1

The Markings On Mars As The Planet Rotates.
Photographs taken in 1939 by E.C. Slipher, Lowell Observatory

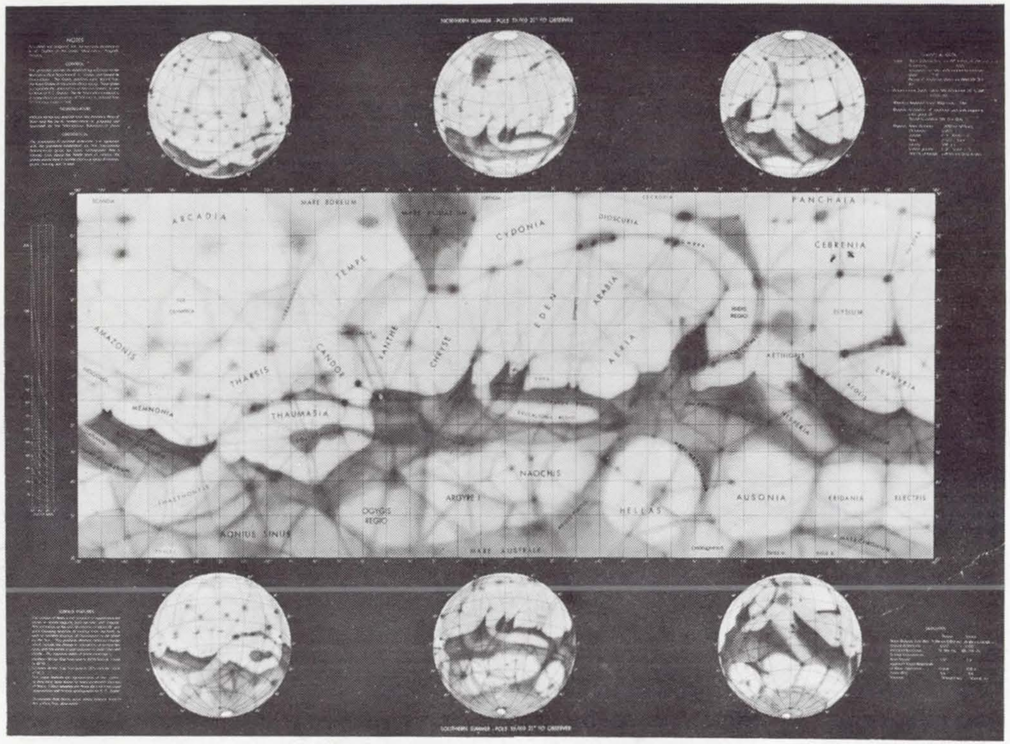

FIGURE 2

Map of Mars as prepared on a special project by the United States Air Force. Follows primarily the Lowell Observatory maps.

FIGURE 3

Photograph of the Painted Desert east of Flagstaff, Arizona. The ground level in the valley is about 5,000 ft. and the elevation of the airplane was about 18,000 ft. Pinon-juniper forests on top of the bluffs (lower left) appear as dark areas as do brushy regions in the bottom of washes. The rest of the desert appears unvegetated, although upon close examination on the ground, no area is completely devoid of plants. Angular marking in the upper left indicates approximate position of the photograph shown in Figure 4.

FIGURE 4

Photograph of a small area of the desert as shown in Figure 3.

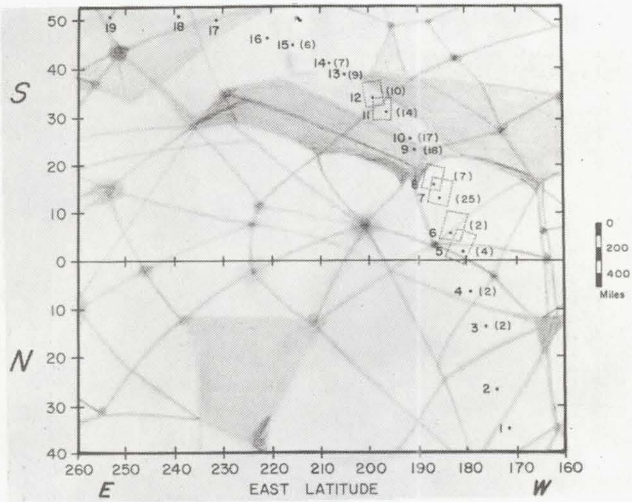

FIGURE 5

Location on a map of Mars of the photographs taken by Mariner IV. Map prepared from one of E. C. Slipher's (Mars, the Photographic Story, Northland Press, Flagstaff, Arizona). Numbers in parentheses indicate the approximate number of craters observable on the photographs. The map is shown according to the convention of the astronomer with south at the top. If the direction of the Mariner camera can be stated with certainty within only one degree, then the location of the photographs on the map can be stated with certainty within an area which may be visualized by mentally displacing the pictures approximately half of their width in any direction (one degree at 8,000 miles represents an arc of 164 miles). It is interesting to note that other maps might show rather significant deviations in the shape and placement of the markings, making the evaluation of the Mariner photographs particularly difficult.

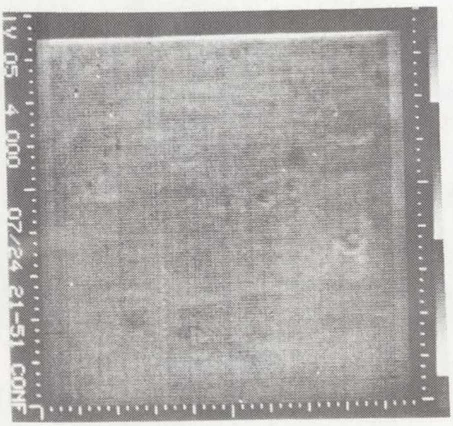

FIGURE 6

Mariner Photograph Number 5

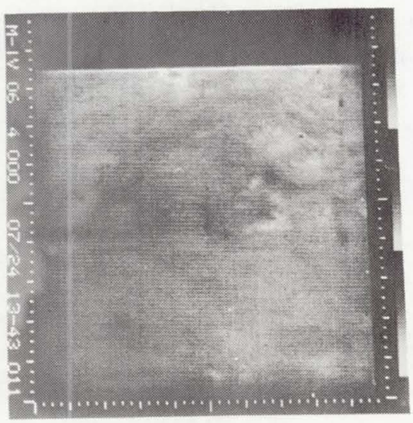

FIGURE 7

Mariner Photograph Number 6

The photographs appear more natural with north at the top so that light is coming from above. The data block on the left indicates that north is at the top.

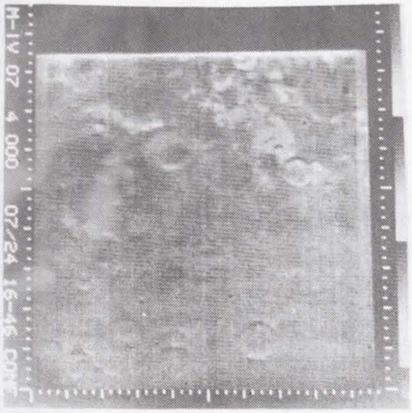

FIGURE 8

Mariner Photograph Number 7.

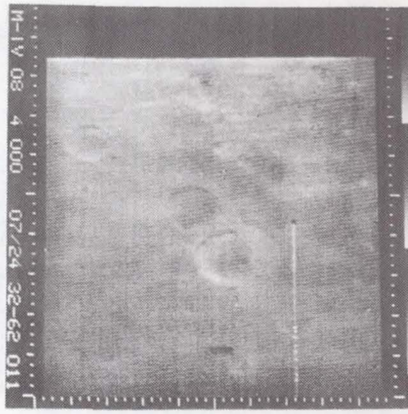

FIGURE 9

Mariner Photograph Number 8

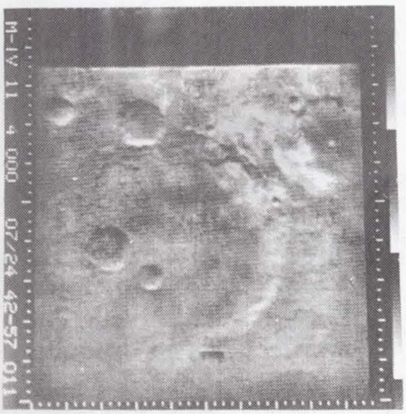

FIGURE 10

Mariner Photograph Number 11.

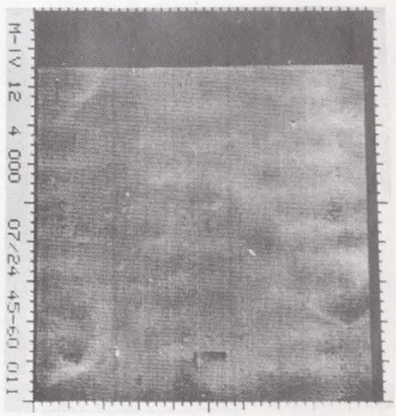

FIGURE 11

Mariner Photograph Number 12.

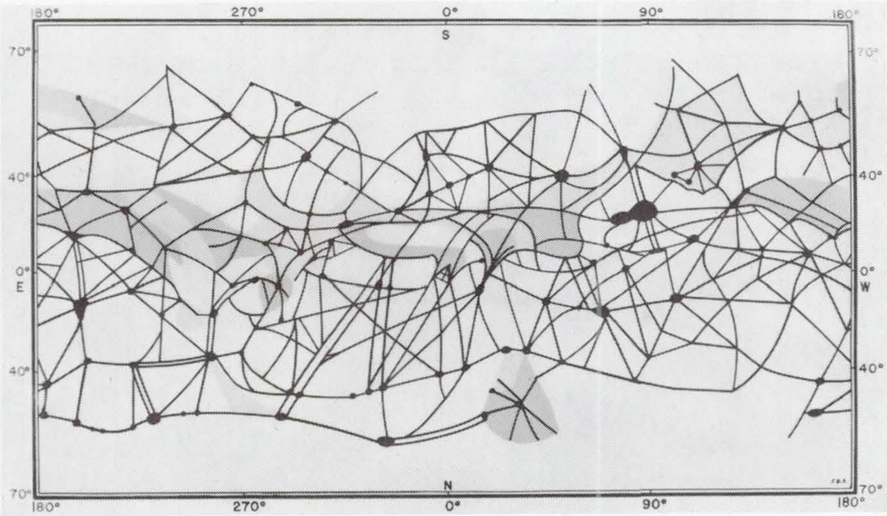

FIGURE 12

The canal network as traced from E. C. Slipher's map. South is at the top. Highly schematic.

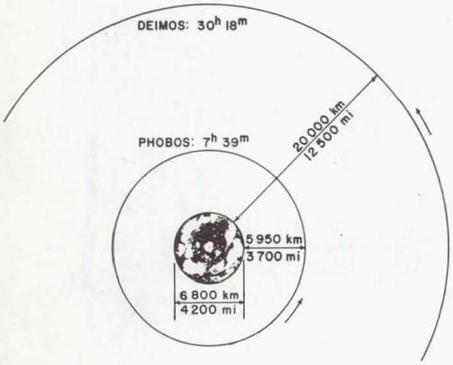

FIGURE 13

Some data relating to the satellites of Mars.

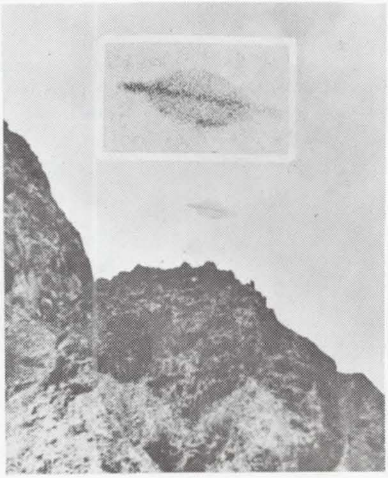

FIGURE 14

The third (and best) photograph of the UFO taken by Almiro Barauna on January 16, 1958 near the Island of Trindad.

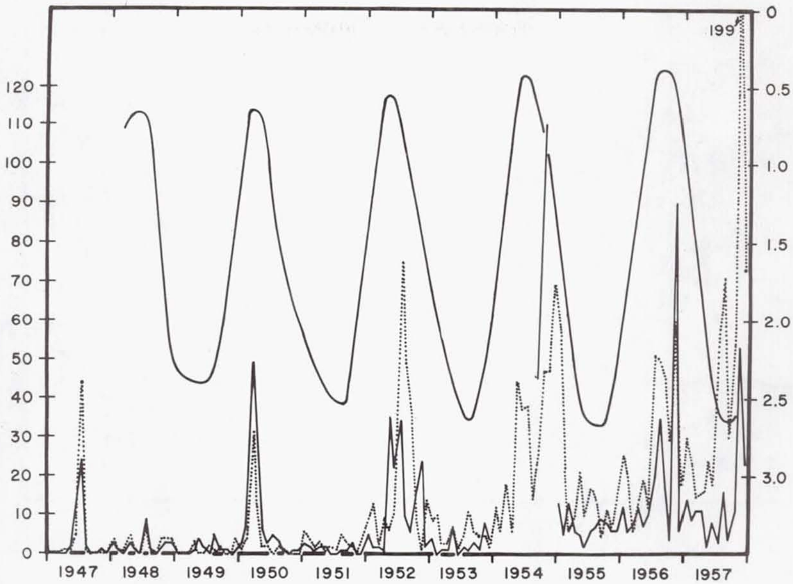

FIGURE 15

UFO sightings in the oppositions of Mars. The dotted line represents "reliable" sightings in the files of the Aerial Phenomena Research Organization, 3190 E. Kleindale Road, Tucson, Arizona. They were supplied by Coral E. Lorenzen. The solid line represents sightings assembled from published reports by M. G. Quincy and presented by Jacques Vallee in The Flying Saucer Review (September, October, 1962). Obviously, there is no correlation between UFO sightings and the distance to Mars in 1947 and 1957, but Vallee calculated correlation coefficients for the other years and found that the observed correlation would be expected to occur due to chance alone less than one time in a thousand trials (all sightings were moved forward two months to account for the lag apparent in 1952, 1954, and 1956).

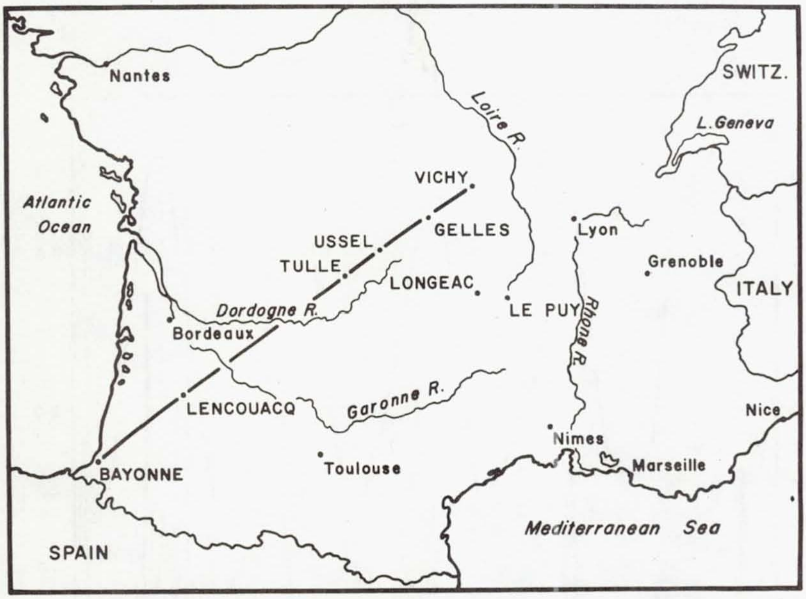

FIGURE 16

Eight sightings in France for September 24, 1964 as assembled by Aime Michel and as reported in France-Soir, Paris-Presse, and La Croix (September 26, and 28). A ninth sighting at Lantefortainles-Baroches in northern France is not shown on the map. Sightings at LePuy and Langeac do not occur on the line, but the other six fall so close to the great-circle arc that no deviation can be detected on a Michelin map with a scale of 1:1,000,000. Circumstances of the six sightings on the line were very briefly as follows. Vichy, afternoon: Football players practicing in a stadium and spectators saw an elliptical, cigar-shaped object cross the sky swiftly and silently. Gelles, early night: The witnesses saw a luminous, cigar-shaped object cross the sky at fairly high speed and without noise. Ussel, about 11 p.m.: A luminous red object rose above the horizon and dived, at high speed, toward M. Cisterne who was driving his tractor back to the barn. The object approached so closely that M. Cisterne jumped from the tractor and lay terrified in the field. The object hovered a few yards above the road, and in front of the tractor, remaining motionless for several minutes in complete silence. Surroundings were illuminated with a reddish light. The UFO then flew over the tractor and disappeared over the horizon in a few seconds. Two other people also saw the object, and leaves at the top of an ash tree near where the object reportedly had hovered, were dried and curled. Tulle, 11 p.m.: M Besse, with the aid of high-powered binoculars, watched a luminous object move rapidly in the sky changing color from reddish to white and then to green. Lencouacq, nightfall: A single witness watched a luminous object arrive at high speed in silence, hover above a meadow, and then leave again at high speed. Bayonne, afternoon: Many people watched three elliptical objects, metallic in appearance, hover in the sky, and then move away very rapidly.

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CHEMICAL STUDIES ON THE ORIGIN OF LIFE

By

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The Space Science Board of the National Academy of Sciences in an authoritative statement declared that the search for extraterrestrial life was the prime goal of Space Biology. "It is not since Darwin and, before him, Copernicus, that science has had the opportunity for so great an impact on the understanding of man. The scientific question at stake in Exobiology is the most exciting, challenging, and profound issue, not only of the century, but of the entire naturalistic movement which has characterized the history of western thought for over 300 years. If there is life on Mars, and if we can demonstrate its independent origin, then we shall have a heartening answer to the question of unprobability and uniqueness in the origin of life. Arising twice in a single planetary system, it must surely occur abundantly elsewhere in the staggering number of comparable planetary systems."[1]

For systems outside our own planetary system, one way of answering the question is by radio contact. Although in the long run listening for evidence of intelligent life may be a profitable and exciting pursuit, the difficulties encountered may be literally astronomical. We are thus left with two principal approaches to cosmobiology: Earth-bound studies on the origin of terrestrial life, and the exploration of other planets to determine whether life has also occurred there. Since the laws of chemistry and physics, presumably, are valid all over the universe, the recapitulation in the laboratory of the path by which life appeared on earth would give strong support to the theory of its existence elsewhere in the universe. Laboratory experiments on earth can reveal which material and conditions available in the universe might give rise to chemical components and structural attributes of life as we know it.

Three factors have made the scientific approach to this question possible, not only theoretically but also experimentally: astronomical advances, recent progress in biochemistry, and the triumph of Darwinian evolution. Present day telescopes reveal that there are more than 10^{20} stars. Therefore, there are more than 10^{20} opportunities for the existence of life. A conservative estimate made by Harlow Shapley suggests that of these at least 10^{8} are suitable for life.[2] Su-Shu Huang was less rigorous in the restrictions he imposed, and he suggested at least 10^{18} possible sites for the existence of life.[3] Harrison Brown has studied the distribution of stars as a function of their luminosity. In a recent paper, he has suggested that for every visible star, there are 60 unseen bodies larger in mass than Mars. He further estimates that 4.2 per cent of these unseen bodies would receive the right amount of heat suitable for life. According to Brown, in our own galaxy there may be as many as 10^{11} planetary systems.[4] In the light of these observations life may, indeed, be prolific in the universe.

Biochemical research during the last decade has established the remarkable unity of biochemistry. In all living organisms, from the smallest microbe to the largest mammal, there are two basic molecules: the nucleic and proteins. While each one of them is complex in form, the units comprising them are few in number. The nucleic acid molecule consists of nucleotides strung together like beads along a chain. The nucleotides, in turn, are made up of a purine or pyrimidine base, a sugar, and a phosphate. In the protein molecule, 20 amino acids link up with one another to give the macromolecule. We are thus led to the inescapable conclusion that all life must have had some common chemical origin.

The Darwinian theory of evolution has postulated the unity of the earth's entire biosphere. According to Darwin, the higher forms of life evolved from

the lower over a very extended period in the life of this planet. Fossil analysis has shown that the oldest known forms of life may be about 2-1/2 billion years old. Life, indeed, had a beginning on this planet. The consideration of biological evolution leads us to another form of evolution, namely, chemical evolution.

The scientific thinking of this problem was crystallized during the first half of this century, especially through the efforts of Oparin,[5] Haldane,[6] and Bernal.[7] Oparin postulated a primitive reducing atmosphere in which a large amount of organic material accumulated before the origin of life and a long chemical evolution as a necessary preamble to the origin of life. Haldane suggested the idea of the "primordial soup" which consisted of an ocean of organic matter which gradually gave rise to replicating systems. Bernal described the methods by which small molecules that may have been synthesized could have concentrated in lagoons or clay deposits by the sea. An accumulation of organic matter was a necessary prerequisite for chemical evolution.

During the last decade, several experiments in this field have established the possible synthesis of molecules of biological significance under simulated primitive Earth conditions. Notable among this work is the classical experiment of Stanley Miller,[8] who, in 1953, exposed a mixture of methane, ammonia, water, and hydrogen to an electric discharge and obtained amino acids and organic compounds like urea, formic acid, etc.

The work I am about to describe concerns recent investigation conducted in the Exobiology Division of the NASA Ames Research Center, Moffett Field, California.[9] The simple working hypothesis which we have adopted is that the molecules which are fundamental now were fundamental at the time of the origin of life. We are investigating the synthesis of the constituents of the nucleic acid molecule and the protein molecule. We simulate primitive earth conditions, prepare the "primordial soup" described by Haldane, and then we proceed to analyze it (Fig. 1).

A starting point for any such experimental work must center around cosmic abundances. Astronomical spectroscopy reveals that the most abundant elements in our galaxy are in the order of rank: hydrogen, helium, oxygen, nitrogen, and carbon. Hydrogen, oxygen, nitrogen, and carbon are indeed the basic elementary constituents of all living organisms. We know from chemical equilibria, that in the presence of hydrogen, carbon, nitrogen, and oxygen must exist in their reduced forms as methane, ammonia, and water. The equilibrium constants for these reactions at 25°C are all of considerable magnitude. It is this atmosphere of methane, ammonia, water vapor, and small amounts of hydrogen which we shall consider as the primitive atmosphere of the earth.

The energies available for the synthesis of organic compounds under primitive earth conditions are ultraviolet light from the sun, electric discharges, ionizing radiation, and heat. While it is evident that sunlight is the principal source of energy, only a small fraction of this was in the wavelength below 2000A, which could have been absorbed by the methane, ammonia, and water. However, the photodissociation products of these molecules could absorb energy of higher wavelengths. Next in importance as a source of energy are electric discharges such as lightning and corona discharges from pointed objects. They occur close to the earth's surface and, hence, would more efficiently transfer the reaction products to the primitive oceans. A certain amount of energy was also available from the disintegration of uranium, thorium, and potassium 40. While some of this energy may have been expended on the solid material, such as rocks, a certain proportion of it was available in the oceans and the atmosphere. Heat from volcanoes was another form of energy that may have been effective, but in comparison to the energy from the sun, this was only a small portion and perhaps not widely distributed.

In our experiments with ionizing radiation, we have found that the electron beam from a linear accelerator at Lawrence Radiation Laboratory of the University of California, Berkeley, provided us with a convenient source of electrons simulating K^{40} on the primitive earth (Fig.2). When a mixture of methane, ammonia, and water was irradiated with 4 1/2 mev electrons for a period of about one hour, resulting in a total dose of approximately 7×10^{10} ergs, and the resulting material analyzed, the largest single nonvolatile compound formed was adenine.[10] The production of adenine in this experiment was significant in the light of the multiple role played by adenine in biological systems. Not only is it a constituent of both DNA and RNA, but it is also a unit of many important cofactors.

In our experiments simulating lightning on the primitive earth, we used 4 tesla coils which were discharged in a 5-liter flask at a voltage of 40 to 50 Kv. Mass spectrometric analysis showed that at the end of 24 hours over 95 per cent of the methane had been used up. At this point, the gas phase consisted mainly of hydrogen cyanide and hydrogen.

When a mixture of methane and ammonia in the presence of water vapor is passed through a heated vycor tube at about 1000°C and the effluent gases absorbed in water, amino acids are formed. This result has recently been reported by Fox, who identified 14 of the amino acids commonly present in protein, in a single experiment.[11] Analysis of the gas fraction has shown that a great portion of the methane is converted into higher hydrocarbons, including ring compounds such as benzene, tolune, and anthracene.

Chemosythesis by meteorite impact on planetary atmospheres has been suggested as a possible pathway for primordial organic systhesis.[12] The reaction is probably a result of the intense heat generated momentarily in the wake of the shock wave following the impact. In a very preliminary experiment simulating these conditions, by firing a ballistic missile into a mixture of methane, ammonia, and water vapor, we have been able to identify some amino acids and a few uv absorbing compounds which may be of biological significance.

When an aqueous solution of hydrogen cyanide, approximately 10^{-3} molar in concentration, is exposed to uv, a wide variety of organic compounds can be formed. Among these have been identified adenine, guanine, and urea. Adenine and guanine are the two purines in RNA and DNA. Urea is an important chemical intermediate. The reaction with hydrogen cyanide may proceed even without a source of energy. When an aqueous solution of hydrogen cyanice is left standing at -10°C, it appears to be converted spontaneously into more complex organic compounds.

In experiments starting with formaldehyde in a very dilute aqueous solution, the two sugars, ribose and deoxyribose, have been identified. These two are the only sugars in RNA and DNA.

The experiments described have shown us that primary molecules or micro molecules could have arisen on the earth devoid of life. The next stage is to find out the conditions under which those molecules could combine together to give more complex arrangements. Such a combination in most cases requires the removal of a molecule of water. When two amino acids are joined together to give a dipeptide, a molecule of water is removed and in so doing, the peptide bond is formed. Similarly, in the formation of a nucleoside, the bases and the sugar are joined together by the elimination of the constituents of the water molecule. In a further stage, when a nucleotide is synthesized by the addition of a phosphate to a nucleoside, elimination of water is once again the prerequisite. Such a reaction may be described as a dehydrogenation-condensation. A condensation of this type can take place either in the water solution, in our case the primordial ocean, or in the relative absence of water, on the shore of the ocean or the dried-up bed of a lagoon.

In our simulation experiments in the laboratory, we have attempted to reconstruct both models. We have been able to demonstrate that the dehydrogenation-condensation reaction can take place under both aqueous and hypohydrous conditions. When a dilute solution of adenine and deoxyribose was exposed to ultraviolet light, we found that deoxyadenosine was synthesized.[13] It was also observed that in such a reaction an organic catalyst can effectively promote the condensation. Several such molecules could have existed on the primitive earth. Among them are hydrogen cyanide and cyanamide. Very striking results were obtained in the case of hydrogen cyanide. This is most significant since hydrogen cyanide is one of the primary products of the action of electric discharges or ionizing radiation on the earth's primordial atmosphere.

A second compound which gave promising results was cyanamide. Cyanamide has been formed when methane, ammonia, and water were exposed to the action of ionizing radiation. Cyanamide is related to hydrogen cyanide in that the hydrogen of the hydrogen cyanide is replaced by a NH_2 group. In attempts to synthesize a dipeptide from the two amino acids, glycine and leucine, we employed cyanamide. When a dilute solution of the two amino acids in the presence of cyanamide was exposed to ultraviolet light, several dipeptides were formed: glycyl-glycine, glycyl-leucine, leucyl-glycine, and leucyl-leucine. It was also noted that some tripeptides were synthesized in this reaction.[14]

The second method of condensation was a heterogenous reaction simulating the dried-up ocean bed. In simulating these conditions, we heated an intimate mixture of the nucleosides with inorganic phosphate, which could have occurred on the primitive earth. Several phosphates were used in this reaction. Among them were disodium monohydrogen, trisodium, sodium ammonium monohydrogen, ammonium dihydrogen, diammonium monohydrogen, monocalcium and tricalcium orthophosphates, and phosphoric acid. When the intimate mixture of the nucleosides the phosphate was heated, phosphorylation took place. The mononucleotides were identified in the end products[15] (Fig. 3). The best yields in this reaction were obtained at about 160°C. However, a small yield was obtained at temperatures as low as 50°C. In the reaction products, we have been able to observe small amounts of the dinucleotide as well.

A study of the mechanisms involved in the production of these biological molecules seems to point out to a simple and straightforward chemistry. In the electric discharge experiments, we noted that at the end of 24 hours all the methane was converted into various organic compounds. In the water soluble fraction we found that 18% of the carbon was present as hydrogen cyanide. Hydrogen cyanide thus appears to be a very important intermediate. Formaldehyde appears to be an important volatile product of the interaction of energy with methane, ammonia, and water. These two molecules in the presence of water can provide us with the building blocks that can go to make nucleic acids and proteins. The hydrogen cyanide, for example, in the presence of ammonium hydroxide condenses with the cyanide ion to give rise first of all to the dimer and then to the trimer. The trimer and the dimer can combine to give adenine, which is the pentamer of hydrogen cyanide.[16] Here we have the pathway for the formation of the purine ring of the nitrogen bases present in the nucleic acids.

A reaction that has been known to organic chemists for over 50 years is the base-catalyzed polymerization of formaldehyde. The first product appears to be glycolaldehyde. The glycolaldehyde gives rise to dihydroxyacetone. These two molecules can combine to give a pentose. The glycolaldehyde also dimerizes to give tetroses. Two tetroses can then combine to give a sugar having eight carbon atoms. By metathesis this molecule can split up into trioses and pentoses. The formation of amino acids from hydrogen cyanide and aldehyde is also recognizable to the organic chemist. The cyanide and aldehyde can give rise to a nitrile. The nitrile is then further hydrolyzed to give rise to the amino acid. The scheme outlined may appear to be oversimplified; however, since every manifestation of life must ultimately be expressed at a molecular level, a working model of the pathways by which methane, and ammonia can give rise to biological molecules of significance can provide us with a useful tool.

Recent developments in quantum biochemistry have thrown new light on the origin of biochemical molecules. The work of Bernard and Alberte Pullman of Paris has highlighted some of the important features.[17] When a quantum chemist takes a first look at the molecules significant in living organisms, he is struck by the remarkable fact that almost all the biomolecules which are essential to living processes are conjugated systems rich in pi electrons. The three fundamental units, the nucleic acids, the proteins, and the energy rich phosphates, exhibit this phenomenon of electronic delocalization. The most significant constituents of the nucleic acid molecule are the purines and pyrimidines. These are conjugated heterocycles. The proteins do not at first sight appear to be conjugated, but the overall structure implies some degree of delocalization. In the helical structure of the protein there is hydrogen bonding between adjacent amino acids. This hydrogen bonding permits a certain amount of electronic transfer. In the case of the energy-rich phosphates, the mobile electrons of the phosphoryl group always interact with the electrons of another phosphoryl group or with the pi electrons of an organic radical. Among other conjugated molecules which are important are the porphyrins. The porphyrin is made up of 4 pyrroles joined together to a central metal atom. The outer structure of the porphyrin is an alternating sequence of double bonds, implying a high degree of conjugation and electronic delocalization. A conclusion that one is obliged to draw from even a superficial consideration such as this is that the basic manifestations of life are intimately connected with the existence of highly conjugated compounds. For some reason, these compounds were "chosen by nature" as the vehicle of life. Electronic delocalization is perhaps the single great quantum effect in biochemical evolution.

Let us for a moment consider how this outstanding characteristic of biomolecules can account for their occurrence as the principal building blocks of living matter. Two major factors appear to be (1) stability requirements and (2) functional advantages. The major result of delocalization is an increment in stability. Quantitatively, this increment is defined as resonance energy. This thermodynamic stabilization must have played an important part in the selection of early molecules. It may be considered to be a period during which there was a struggle for survival. There was a selection of biomolecules and those which finally triumphed were the ones which were more stable. This fact is confirmed by the extraordinary unity of biochemistry which we discussed earlier. The same limited number of compounds performs the same functions over the entire plant and animal kingdom. To take one example, chlorophyll, which is used by plants, is very similar to heam which is used by animals. Both compounds are synthesized by the same sequence of reactions. Stablization by electron delocalization may also have played a part in the orientation of small molecules to give the large polymers such as the nucleic acids and proteins.

In concluding this discussion on electronic delocalization, the following observations may be made: (1) evolutionary selection used the most stable compounds; (2) on account of electronic delocalization these compounds were best adapted for biological purposes; and (3) life did not originate with the appearance of the conjugated compounds, but the possibility of life as we know it was made more probable by their appearance.

Carbon and silicon appear in the same group (IVA) of the periodic table and both need 4 electrons to reach the configuration of the nearest inert gas. On account of this superficial and apparent similarity between carbon and silicon the question of a "silicon biology" has often been raised in discussions on the origin of life. However, a careful consideration seems to indicate that such a prospect is very unlikely.

One answer to the question is forthcoming from a consideration of cosmic abundance. Carbon is certainly more prevalent than silicon in the universe. Another reason arises from the fact that hydrogen, carbon, nitrogen, and oxygen have been utilized in living systems, since they are the smallest elements in the periodic table and can achieve the stability of inert gases by the addition of 1, 2, 3, and 4 electrons. Small atoms form tight and stable molecules. They can also form multiple bonds. In comparison to carbon, silicon forms weaker bonds with itself and other atoms. Silicon does not form multiple bonds, and the result is the formation of large polymers like quartz, which are unwidely and also remove any available silicon from circulation.

Optical activity has often been suggested as a very distinctive characteristic of molecules present in living systems. In living organisms all syntheses and degradations involve only one enantiomorph. While a start with one form or the other would have been self-perpetuating, it is difficult to understand how the initial choice was made. Physical forces, such as circularly polarized light, the surface of asymmetric crystals, or spontaneous crystallization, cannot account for the overwhelming tendency to produce only one form rather than the other. A reasonable explanation appears to be that the structural demands of large molecules required the use of one form rather than both. The use of one optical isomer rather than mixtures would, undoubtedly, confer great stability on the polymers. This still does not answer the question of how the initial choice was made. The most plausible solution appears to be that the single optical isomers were selected on the basis of stability of the structures. It was a case of natural selection at the molecular level.

In a recent paper,[18] Miller has discussed the possibility whether life could exist in solvents other than water. Many solvents have often been suggested, such as liquid ammonia, hydrogen cyanide, hydrogen fluoride, hydrogen chloride, alcohols, hydrocarbons, fused salts, etc. Some of these can be summarily dismissed on the basis of atomic abundance, rapid decomposition, or the instability of organic compounds in them. The only one that merits some consideration is liquid ammonia. On account of the high cosmic abundance of water, liquid ammonia will be contaminated with water, and would give rise to 3 eutectics containing 34 per cent ammonia, 58 per cent ammonia, and 82 per cent ammonia. According to Miller, this system would also be unsuitable for living organisms. The composition of the

solvent would change considerably because of the different volatilities of ammonia and water. The growth of organisms would be inhibited since a constant environment is necessary for living systems. As the solids are more dense than the liquid, the oceans will freeze from the bottom up and the environment will no doubt suffer from this unfavorable circumstance. In conclusion, we might remark that while it is not possible to prove that life cannot arise or grow in nonaqueous media, it can certainly be said that they are unfavorable for the survival and growth of life.

The results of our laboratory experiments and conclusions from our consideration of chemical bonding can be applied to our program on the search for extraterrestrial life. We have learned that molecules of biological significance can be synthesized abiologically. The finding, therefore, of biological compounds on the surface of Mars is no indication that life exists there. Chemical studies on Martian samples will have to be followed by more direct evidence for life, such as metabolism and reproduction. Another fact which becomes clear is that if life is the result of the inevitable evolution of matter, life elsewhere in the universe will be very similar, at least chemically, to ours. When this idea is supported by quantum chemical conclusions, one is inclined to the view that all life is very probably based on carbon and employs nucleic acids and proteins. We would therefore be justified in designing our life detection experiments on the basis of what we know about life on earth.

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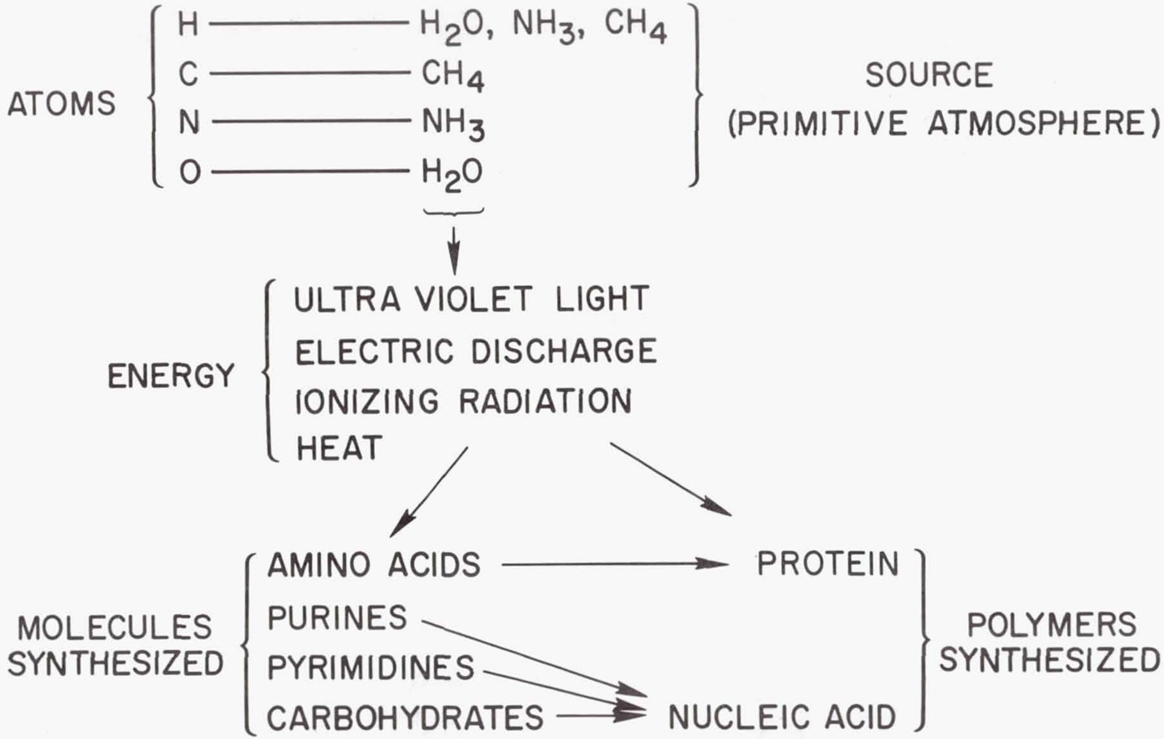

FIGURE 1

Synthesis of Organic Compounds On The Primitive Earth

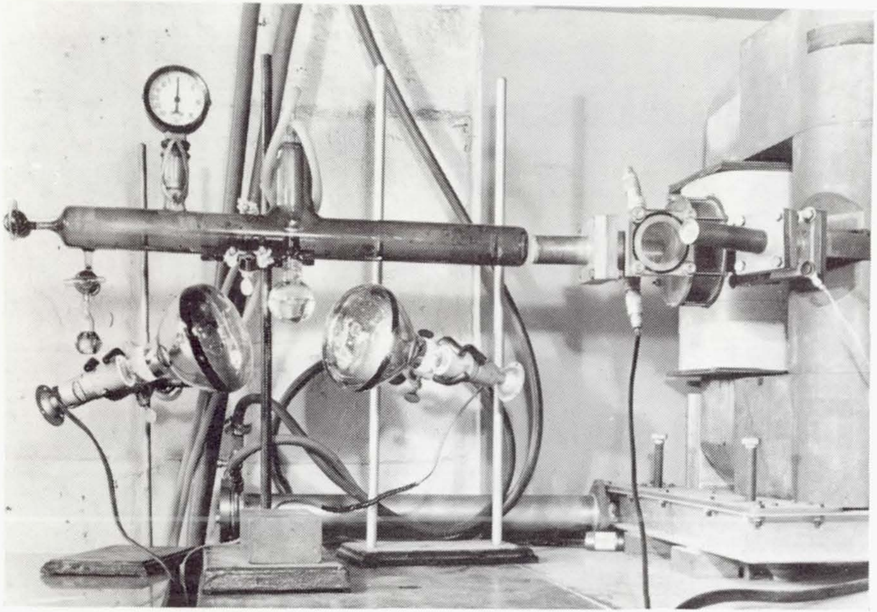

FIGURE 2

Apparatus for the Electron Irradiation of Methane, Ammonia, and Water Using 5 Mev Electrons from the Linear Accelerator of the Lawrence Radiation Laboratory, Berkeley, California.

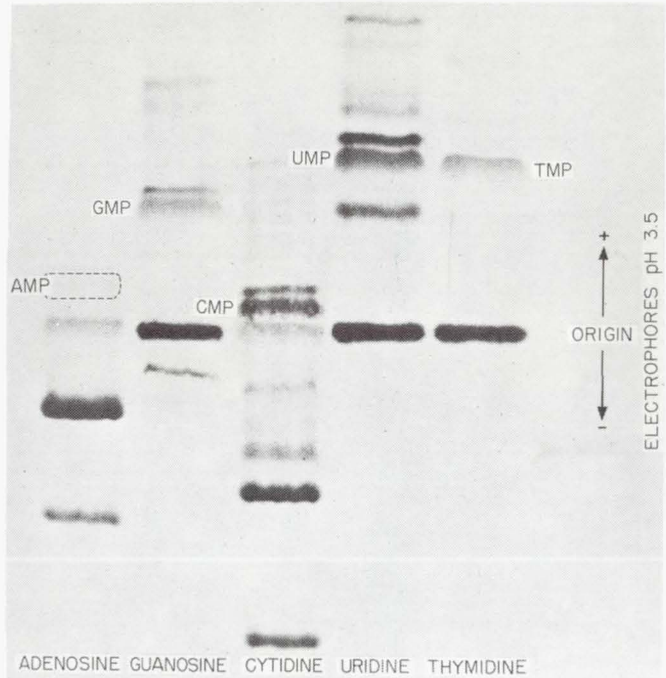

FIGURE 3

Autoradiograph showing systhesis of mononucleotides

SPECIAL ORBITS FOR THE EXPLORATION OF MARS AND VENUS

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INTRODUCTION

The thesis of this paper is that since the amount of available propulsion and power for space oriented missions is finite, the proper utilization of these is essential not only for the success but also for the possibility of certain missions. The finding of the "best" utilization of the fuel is known as a problem in optimization research. This optimization process, however, can never proceed without first selecting the general type of trajectory for a given mission. In other words, optimization techniques often furnish only local optima in the sense that once the basic type of orbits available are characterized, then the "best" ones might be found by the various techniques of trajectory optimization.

The first kind of trajectory of interest here is called the swing-by trajectory, the second kind is known as capture-orbit. The first kind may be regarded as a dynamical idea utilizing the gravitational forces of the bodies of the solar system to accomplish a mission. The existence and the usefulness of such swing-by solutions is not obvious. Numerical experiments, however, verified the expectations and these types of trajectiories have significant advantages over those which may be termed conventional.

The problem of the existence of capture-orbits is much more difficult since in this case the question is directed at the long-time behavior of orbits and purely numerical methods seldom answer such questions.

My purpose is to call attention to the two unusual trajectory problems and to describe a numerical method for treating these with high accuracy. Detailed results as well as preliminary orbits are available regarding families of swing-by trajectories (ref 1., 9.). Several highly competent and carefully performed optimizations associated with these orbits are also in the literature. Consequently, I will only give a few examples regarding the general idea and try to stress the underlying analytical, dynamical, and numerical aspects.

SWING-BY TRAJECTORIES

One of the fundamental problems in research in the field of interplanetary trajectories may be formulated as follows:

There are at least two points P_1 and P_2 given inside a sphere of radius, say 50 a.u. $\sim$ 7500 x 10^6 km, with associated times when the probe must occupy the neighborhood of these points. Within the framework of the constraints imposed on the problem, find the "best" orbit between P_1 and P_2. Here "best" usually means an orbit which is associated with minimum fuel consumption. Note that one and often both of the previously mentioned times are given only within certain limits instead of with specified precision and that usually only the duration between these times is specified. Consequently the "time" is specified only in a relative sense and because of the non-autonomous nature of the dynamical system, the critical space-coordinates of points P_1, P_2,...may also not be given in an absolute sense. Sometimes the points P_1, P_2,...are given in the phase-space, that is the velocity of the probe is also prescribed at these points.

Before we leave this general formulation of the problem, the following comment regarding the sphere of radius R_o = 7.5 x 10^9 km is in order. The dynamically significant natural members of the solar system; the sun, the planets, their satellites, the asteroids and the comets, as well as some of the artificial bodies, such as the planetary probes, with almost no exception describe orbits which lie in the general vicinity of the ecliptic. Without referring to possible cosmogonical explanations of this fact regarding the natural bodies, we mention the well known reason for not

finding man-made probes outside of the ecliptic. The velocity available for launching a probe is a combination of the velocity of the launching platform (a planet, in fact the Earth) and the velocity produced by the propulsion device. Since the direction of the first component of the velocity is in the ecliptic plane, the orbit of an artificial body deviating significantly from this plane must be produced by a large velocity component, normal to the motion of the platform. In other words, orbits having large inclinations to the plane of the ecliptic are "expensive" in terms of fuel. We shall see in what follows that swing-by trajectories may also accomplish out-of plane missions without much additional fuel. So while conventional missions are restricted to a circular area of radius R_o, the idea of swing-by adds another dimension and geometry of operations becomes three-dimensional within a sphere of radius R_o.

In order to characterize swing-by trajectories a standard round-trip mission to Mars as shown in Figure 1 is compared with the swing-by mission shown in Figure 2. The corresponding points in these two figures are numbered from 1 to 7 and the data and velocities are given in Table 1. These figures and the associated data have been prepared by Mr. Deerwester (see reference 1) whose kind permission for their use is most gratefully acknowledged.

Observe that the duration of the mission is considerably longer in the swing-by mode than in the standard operation in spite of the same stop-over time at Mars. This is generally a consequence of operating in the swing-by mode.

Another interesting comparison which may be made using Table 1 is related to the departure velocities from Mars. It is important that this is half as large for the swing-by mode as for the standard operation.

One of the most significant effects of the swing-by mode is that the speed with which the probe arrives at the Earth is significantly reduced. This is important since one of the most critical items of interplanetary round-trip missions is the re-entry velocity of the returning probe to the Earth's atmosphere. As the table shows the re-entry speed is greatly reduced by the swing-by mode of operation.

The dynamics of the swing-by operation may be explained by a comparison with a standard trajectory. When the probe is on a standard trajectory its orbit is determined by the home-planet, by the target-planet and by the Sun. In fact its elliptic orbit governed by the Sun is perturbed significantly only at the beginning and at the end of the trajectory by the home and target planets. Other perturbations, mostly by planets not participating directly may be ignored in this first approximation.

On the other hand, operation in the swing-by mode calls for a large perturbation of the heliocentric elliptic orbit. This perturbation is a three-body effect since the perturbing planet (which produces the swing-by) forms a system of three bodies with the Sun and the probe. These three bodies have the important special property that the mass of one of the bodies (the probe) is much smaller than either the mass of the governing body (the Sun) or that of the perturbing body (say, Venus). Consequently, no matter how closely the probe approaches the perturbing planet, no effect of the probe on the orbit of this planet is observed. This fact is important and it simplifies the computation of the perturbations on the probe.

Another dynamical consequence is that the problem of three bodies mentioned before may be split up into two problems of two bodies under the circumstances prevailing during a swing-by operation. Prior to the close approach of the probe to the perturbing planet, the probe's orbit is governed by the Sun, provided that the probe has departed sufficiently far from the home planet. The two-body problem formed by the Sun and the probe may be solved and in this way the probe's Sun-centered elliptic orbit may be obtained. This will be only an approximation to the actual orbit since all perturbations are neglected. As the probe approaches the planet which is to cause the swing-by, its orbit deviates from the previously mentioned two-body orbit and it becomes a so called perturbed two-body orbit. When the probe is in the close vicinity (say within the sphere of action) of the planet, the effect of the Sun as compared to the planet may be neglected in the first approximation and we once again are faced with a problem of two bodies, consisting at this time of the probe and the planet. The energy of the probe relative to the planet is conserved because a two-body system for the planet and the probe was assumed.

Let the heliocentric velocity vector of the planet which is responsible for the swing-by be denoted by $\overline{V}_p$. This vector of course is not constant; its direction and its magnitude both change. During the short time of encounter with the probe, however, we may assume $\overline{V}_p$ to be constant. If the velocity vectors of the probe relative to the planet before entering and after leaving the sphere of action are $\overline{v}_1$ and $\overline{v}_2$, then the velocity vectors of the probe relative to the Sun become

$$\overline{V}_1 = \overline{V}_p + \overline{v}_1$$

and

$$\overline{V}_2 = \overline{V}_p + \overline{v}_2.$$

The energy is conserved in the two-body system formed by the planet and the probe in the vicinity of the planet, consequently

$$|\overline{v}_1| = |\overline{v}_2|.$$

The direction of the relative velocity, of course, changes: $\overline{v}_1 \neq \overline{v}_2$. The change in the direction ϕ may be computed from the equation

$$\cos \phi = \frac{\overline{v}_1 \cdot \overline{v}_2}{|\overline{v}_1| \; |\overline{v}_2|} .$$

The energy of the probe relative to the Sun, on the other hand, does change since

$$\overline{V}_2^{\,2} - \overline{V}_1^{\,2} = 2\overline{V}_p \cdot (\overline{v}_2 - \overline{v}_1) = 2\overline{V}_p \cdot (\overline{V}_2 - \overline{V}_1) \neq 0.$$

This fact is the key to the dynamics of swing-by trajectories. The planet changes the energy of the probe relative to the sun or in other words the energy of the probe on its orbit around the sun is changed by the planet and consequently the probe describes a new orbit around the sun after the encounter with the planet.

The relation between the elements of the orbits of the probe prior to and after encounter is contained in an equation of celestial mechanics known as Tisserand's criterion for the identity of comets (references 2 and 3). This equation, derivation of which from the Jacobian integral of the restricted problem of three bodies is a relatively simple matter, is

$$\frac{1}{a} + 2\,[a\,(1-e^2)]^{1/2} \cos i = C.$$

Here a is the semi-major axis, e is the eccentricity and i the inclination of the orbit of the probe to the plane of the orbit of Jupiter of approximately to the ecliptic. The constant C represents the identity of a comet; in other words, if a, e, and i are observed before and after the comet's encounter with Jupiter then these quantities must satisfy the invariant relation,

$$\frac{1}{a_1} + 2\,[a_1\,(1-e_1^{\,2})]^{1/2} \cos i_1 = \frac{1}{a_2} + 2\,[a_2\,(1-e_2^{\,2})]^{1/2} \cos i_2.$$

The application of this equation to swing-by trajectories is straight forward (since no special importance is attached to the planet Jupiter) as long as the perturbing planet's mean distance from the Sun is used as the unit of length in the calculations.

An interesting application of the preceeding equation is to the changes in the orbital plane of the probe. Let $i_1 = 0^\circ$ and $i_2 = 90^\circ$; an extreme case indeed, when the original orbit of the probe is in the plane of the ecliptic and we wish to effect a swing-by operation such that after encounter with the perturbing planet the orbital plane of the probe is perpendicular to the plane of the ecliptic. The equation then gives

$$\frac{1}{a_2} - \frac{1}{a_1} = 2\,[a_1\,(1-e_1^{\,2})]^{1/2},$$

from which it follows that

$$a_2 < a_1 .$$

That is, the semi-major axis of the orbit perpendicular to the plane of the ecliptic is smaller than the semi-major axis of the original orbit in the ecliptic. In fact for orbits originally having a small eccentricity we have

$$a_2 = \frac{a_1}{1 + 2a_1^{3/2}}$$

It is to be noted that Tisserand's equation is an approximation to the actual physical situation and therefore its application to swing-by trajectories should be for verifying and checking rather than establishing precise results.

Dr. Stanley Rose kindly gave me the following numbers to demonstrate the use of Tisserand's criterion and to check calculations related to swing-by trajectories. These numbers were obtained by simple two-body calculations using matched conic sections, consequently increased significance is attached to the check offered by Tisserand's criterion. Both orbits, prior to and after encounter are assumed to be in the plane of the ecliptic, so $i_1 = i_2 = 0$. The original orbit around the Sun is an ellipse with a semi-major axis $a_1 = 0.774$ a.u. and with an eccentricity $e_1 = 0.277$. The probe departs from the Earth on 28 November 1978, then executes a swing-by at Venus on 7 May 1979, and arrives at Mars on 16 October 1979. The departure velocity with respect to the Earth is 5 km/sec and with respect to the Sun is 25.8 km/sec. The distance of the closest point of the probe's orbit from the center of Venus is 8250 km. The probe therefore passes above the surface of Venus at a distance corresponding to 0.35 times the radius of the planet. The asymptotic velocity of the probe with respect to Venus increases to approximately 14 km/sec. After the close passage the orbital elements are $a_2 = 1.134$ a.u. and $e_2 = 0.407$. In the application of Tisserand's criterion dimensionless values of the semi-major axes are used with the unit of length being Venus' mean distance from the Sun. The preceeding values of a_1 and a_2 refer to the Earth's mean distance from the Sun as the unit of length, consequently the values to be used in Tisserand's equation are

$$\overline{a}_1 = a_1/.723 = 1.0705$$

$$\overline{a}_2 = a_2/.723 = 1.5684.$$

Evaluating the quantity

$$1/\overline{a}_i + 2\,[\,\overline{a}_i\,(1-e_i^{\,2})]^{1/2}$$

with $i = 1$ gives 2.92245 and with $i = 2$ we have 2.92549; consequently the deviation is approximately $0.1^\circ/_\circ$.

Tisserand's equation assumes circular orbits for the planets around the Sun and planetary masses that are small with respect to the Sun. These assumptions are better satisfied for Venus than for Jupiter since the eccentricity of Venus' orbit is only 0.00679 while Jupiter's is 0.04843. The values of the mass ratios also favor Venus in a ratio of 391 to 1.

Some additional remarks on swing-by trajectories are of interest. The first one concerns the close approaches. The smallest distance of the probe from the planet during the swing-by maneuvers is one of the critical parameters of the mission. For this reason it is not only feasible but in fact advisable that manned probes be considered for swing-by missions. In this way the final adjustments in the close approaches become more flexible. But whether manned or unmanned probes are being considered the effect of the close approach on the subsequent trajectory must be computed with great accuracy. A method to accomplish this is discussed in the third part of this paper.

The question of the guidance requirements comes up naturally at this point since the feasibility of swing-by missions might depend on the availability of accurate guidance methods and equipment. We know today that the guidance requirements of swing-by trajectories do not exceed those for the Apollo mission. This result follows from preliminary numerical work and as such it is subject to further detailed analysis. Nevertheless at the present time it is established that one of the most critical parts of a swing-by mission is still the reentry into the Earth's atmosphere as far as guidance is concerned.

The insufficient accuracy with which planetary radii and the astronomical unit are known today seems to be one of the most serious difficulties in actual flights. Improvements in our knowledge of the scale of the solar system and the size of the planets are definitely needed for the successful performance of swing-by missions. It should be noted at the same time that such missions will furnish information regarding these physical constants of the solar system.

The availability of planets for swing-by operations, depends upon the synodic periods of the planets since once the time for a configuration of the planets is found which makes the swing-by mode possible, other times follow periodically. The discovery of a proper launching time is not a trivial matter and sometimes careful search extended to several years is necessary to establish an efficient swing-by mission. Assume now that one solution is known and let us investigate the repeatability of the operation. If the mean motions (the angular velocities) of two planets are n_1 and n_2, then the angular velocity of their relative motion is

$$n_{12} = n_1 - n_2$$

and consequently the period of the relative motion T_{12} may be computed from

$$\frac{1}{T_{12}} = \frac{1}{T_1} - \frac{1}{T_2} ,$$

where T_1 and T_2 are the sidereal periods of the planets, $T_i = 2\pi/n_i$.

When one of the planets to which the preceeding equation is applied is the Earth, the relative period is called the synodic period.

The synodic period of Venus is approximately T_{EV} = 19.2 months and that of Mars is T_{EM} = 25.6 months, since T_V = 0.6156 year and T_M = 1.8822 year. The system consisting of Venus, Earth and Mars possesses a period of T_{VEM} = 6.4 years which is obtained from the equation

$$\frac{1}{T_{VEM}} = \frac{1}{T_{EV}} - \frac{1}{T_{EM}} .$$

Table 2 gives the approximate sidereal and synodic periods of some of the planets in months. The synodic periods approach 12 months as the sidereal periods increase to large values and $T_{syn} = 0$ when $T_{sid} = 0$.

Note that the eccentricities and inclinations of the planetary orbits are not zero. This fact renders the use of these equations and of the table approximate. Nevertheless, the corrections needed are small and certainly negligible in preliminary computations.

Repeated use of the same planet for swing-by constitutes another possibility. In this case the mean motion of the modified orbit of the probe must be commensurable with the mean motion of the perturbing planet.

The amount of deviation of the orbits due to the swing-by operation depends, of course, on the closest distance between the planet and the probe and on the mass of the planet. Consequently the use of the Earth's moon for swing-by operations is excluded while Jupiter is considered an excellent candidate especially for solar probes.

Preliminary analysis shows that fuel savings are almost always associated with swing-by operations but that swing-by orbits generally take more time than the standard operations. For instance, the mission to Mars and return discussed at the beginning of this section requires 540 days with a swing-by and only 430 days with a direct mode of operation. The fuel saving using the swing-by mode, on the other hand is about 30%.

We add the fascinating preliminary result that with planet-to-planet swing-by maneuvers one may be able to pass near to all the major planets on a single trip.

The references numbered (4) to (17) discuss swing-by missions in considerable detail. Several references describing Mariner missions are also included to facilitate comparisons.

CAPTURE ORBITS

One of the most complex problems of celestial mechanics is the capture problem. There exist several possible formulations of this problem which, in spite of its practical significance, has only been solved approximately for certain special cases.

First the importance of the capture problem is demonstrated with some examples. Based on these examples we then formulate a specific capture problem and present two approximate limiting solutions.

One of the classical capture problems in the cosmogony of the solar system is the origin of the Earth-Moon system. Assume that the Earth either with its present mass or with a different mass, is revolving in its present or in a different orbit with small eccentricity around the Sun. In this system of two bodies a third body of much smaller mass (the moon) is introduced. The question to be answered is whether there exist initial conditions for the moon so that it will become a permanently satellite of the Earth. As long as there are no restrictions on the initial condi-

tions, the answer is in the affirmative if the assumptions and the formulation of the restricted problem of three bodies are accepted. All that is required is that the initial conditions put the moon in an orbit which is inside the smaller oval of the curve of zero velocity surrounding the earth. If the possibility of collision is to be excluded then the problem becomes slightly more difficult. The answer is trivial of course under the given condition since it says in effect that the Earth will keep the moon captured once it is in a capture orbit. Once we specify the initial conditions of the moon differently, the problem becomes of the greatest difficulty. For instance, we might inquire whether capture is possible if the orbit of the moon is initiated in the immediate vicinity of the Sun, or again if the moon is in a planetary orbit around the Sun at an orbitrary distance from the Earth's orbit, etc.

Another capture problem is closely connected with establishing artificial satellites of the bodies of the solar system. Consider the Earth-Moon system and assume that no perturbations are acting on this system. Also assume that the orbits of the Earth and of the Moon are circles around the center of mass. Then, we might ask if it is possible to select initial conditions for a space probe close to the Earth such that it becomes a permanent satellite of the Moon. It is, of course, assumed that only gravitational forces are acting on the vehicle after its initial conditions are established.

A similar question is also of great interest regarding the planetary system. Consider for the purpose of demonstration again a system of three bodies: the Sun, the space probe and a planet. Only one planet is included and the effect of all the other planets is ignored. The pertinent question is whether there exist initial velocity vectors from arbitrary points in the solar system such that a space probe leaving from these points with the given velocities become permanent satellites of the planets.

The affirmative answers to these questions would allow the establishment of the orbits of planetary probes which, with minimum guidance would furnish artificial satellites of the various planets. It is essential to emphasize that the idealization of the problem from the actual physical situation to a simplified model such as the restricted problem of three bodies does not mean that the answer has "only" theoretical interest. Simplification allows finding a solution which approximates the solution of the actual physical problem. This approximate solution requires slight modifications and also possibly the introduction of guidance forces. Nevertheless, these are small corrections which only modify the solution of the simplified system.

One of the powerful methods for studying the capture problem was suggested by Hill, and it consists of the use of the curves of zero velocity in the restricted problem of three bodies (reference 3). Unfortunately this method is designed to answer a question which in some respect is the opposite of the capture problem. Hill's method finds whether a body once in a satellite orbit will remain there; in other words, will a planet lose a satellite? Without discussing curves of zero velocity in any detail we characterize them as constant relative energy curves which a body under certain circumstances may not cross.

Another more general concept applicable to the study of the capture problem is the ergodic theorem. This theorem attempts to determine whether a dynamical system reaches every point of the phase-space. If the system covers its energy surface densely, then capture is not possible. Recently this theorem has been sharpened for the restricted problem of three bodies by numerical and analytical methods so that we may speak about conditionally ergodic system which under certain circumstances show ergodicity. The ergodic theorem of course must be looked upon as a statement in probability theory since a body might perform satellite type motion for a very long but finite (not infinite) time before it leaves the planet.

Closely associated with the ergodic property is the concept of recurrence. If an initial condition repeats itself within a sphere of radius Σ infinitely often, the sphere being located in the phase space, then the system is recurrent. Now, according to Poincaré's cycle theorem, if the phase space is bounded and the dynamical system consists of material particles under their mutual gravitational fields then the system is recurrent. But if this is so then, of course, capture in its most general sense is impossible since if a probe is located "far" from a planet at one time then it will have to return infinitely often to this "far" location as time approaches infinity. The crucial part of Poincaré's cycle theorem is that the phase space must be bounded for its applicability. Even if we restrict the consideration to finite space, the velocity of colliding particles becomes infinite and consequently the phase space is not finite. The cycle theorem is applicable after regularization of the equations of motion in that part of the regularized phase-space where there are no singularities but this restriction unfortunately allows the drawing of trivial conclusions only.

After these, unquestionably discouraging, considerations we formulate two distinct problems of capture. Both formulations use the restricted problem as model and both define capture by means of Tisserand's sphere of action (reference 2). The first capture problem assumes that the initial position of the probe is in the neighborhood of the primary with the larger mass, the second problem allows arbitrary initial conditions. Regarding the possibility of capture Fesenkoff (reference 18) and Egorov (reference 19) have given negative answers in the case of the first problem and a conditionally negative answer for the second problem. We note that there is a certain amount of contradiction in the two answers. The reason may be found in the rather crude set of assumptions used in the derivations.

The sphere of action has the radius 66,000 km around the Moon in the problem including the Earth, the Moon and the probe. It is convenient to define capture by means of this sphere. If the probe at one time is outside of this sphere and at a later time and for all subsequent times is inside, then we speak about capture.

In what follows the reader's patience is requested regarding the use of the seemingly double negative constructions. Most of the results refer to the impossibility of capture. If one cannot show that capture is impossible that does not mean that capture is possible. It simply means that we cannot make any comment regarding the capture-problem. If one can show that capture is possible or that it is impossible then the problem is solved.

Fesenkoff's and Egorov's findings show that free trajectories starting in the neighborhood of the Earth, on their first approach after piercing the sphere of action around the Moon will always emerge. In fact if the mass-ratio of the two participating primaries is smaller than the mass-ratio of the Earth-Moon system ($\sim 1/81$) then capture is always impossible, provided the initial position of the probe is sufficiently close to the larger primary.

These results state that by launching a probe from the vicinity of the earth, capture by the Moon may not be accomplished, whatever the initial conditions are. If a probe is launched from a point sufficiently close to the sun, it may not become a permanent satellite of any of the planets, whatever the initial conditions are. It is important to remember that these results are only as good as the assumptions and the model.

Regarding the second formulation, when the probe approaches the primary on an arbitrary trajectory the impossibility of the capture may be shown only for very small values of the mass-ratio. To give as precise meanings to these results as possible we introduce the mass-parameter defined by

$$\mu = \frac{m_2}{m_1 + m_2}$$

where m_1 and m_2 are the masses of the participating primaries, the mass m_1 being the larger.

In terms of this parameter, I conclude from Egorov's result that capture is impossible if $\mu < \mu_{crit} \sim 6 \times 10^{-5}$. In view of this we may investigate possible capture orbits of probes to the various planets. Table 3 shows the approximate values of the mass parameters when the planets form restricted three-body problems with the Sun. The impossibility of capture shown in the table in case of the planets Earth and Mars is exceptionally interesting since both planets have satellites.

The mass parameter of the Earth-Moon system is larger than the critical value ($0.012 > 6 \times 10^{-5}$), consequently the impossibility of capture according to the second formulation does not follow. This means that in the restricted problem of three bodies where the Earth and the Moon are the primaries, if the probe is allowed any initial conditions then the impossibility of capture by the Moon is not proven. If, on the other hand, the initial position of the probe is close to the Earth then the impossibility of its capture by the Moon may be proved.

The result of Table 3 regarding Venus and Mars mean that the impossibility of capture persist, whatever initial conditions are used. These conclusions agree with the experimentally established fact that in order to create an artificial satellite of the Moon or of any of the planets (other than of the Earth) free trajectories must be modified by midcourse or terminal propulsion. By experiments we do not necessarily mean actual space flights but rather numerical experimentations on computers.

The establishment of temporary capture without additional thrust has been demonstrated by Egorov (reference 19), Thuring (reference 20), and Arenstorf (reference 21). In this case a satellite orbit is established around one of the primaries. This orbit, after several revolutions, switches over to the other primary and the probe becomes a satellite of this primary for awhile. The switching back and forth continues, in fact motion in a periodic fashion is also possible. The problem, however, is not one of capture.

A thorough review of the literature on capture is offered in reference 22.

COMPUTATIONAL PROBLEMS

The two preceeding sections discussed orbits which connect the neighborhoods of members of the solar system. In as much as the Newtonian gravitational force is inversely proportioned to the square of the distance between participating bodies, whenever this distance approaches zero the forces become infinite. We speak of singularities of the differential equations of motion. These are located at the centers of the bodies.

While singularities create analytical problems, close approaches are associated with serious computational difficulties. In other words, it is not necessary to have collisions in the mathematical sense of singularities in order to face serious losses of accuracy during numerical integration.

Both the swing-by trajectories and the capture orbits have the common property that they connect the neighborhood of the singularities. Both trajectories require especially high accuracy for operational and theoretical reasons. Instead of discussing various techniques of numerical integration or describing methods of solution of the differential equations of motion, an approach to the problem is suggested which only recently has been applied to trajectory computations (reference 23).

This method is called regularization and it consists of the introduction of variables such that the singularities of the differential equations of motion are eliminated. Whenever new variables are introduced the crucial question of efficiency immediately enters. If the transformations involved will result in such a high level of complexity that more accuracy and time is lost because of them than in the original system then the introduction of new variables is not advisable. This question often may be answered only experimentally. The usefulness of the transformations which are recommended in the following few paragraphs have been demonstrated many times.

The common feature of all regularizing techniques is the introduction of a new independent variable S by the equation

$$S = \int \frac{dt}{r}$$

where r is the distance between the bodies, close approaches of which might be expected to become critical. This step may be generalized to the transformation

$$S = \int \frac{dt}{f(r)}$$

or to transformations effecting also the dependent variables. The proper choice of these transformations will increase the accuracy and the efficiency of the computations but the essential aspects are included in the first equation, $dt/ds = r$.

The purpose of this paper is not to derive the regularized equations of motion which are available in several references (22, 23), but to call attention to a computational method which is essential to the subjects treated in the two previous sections.

The characteristic features of the two types of orbits discussed in this paper are that preliminary calculations allow very crude approximations, but final results of high precision require sophistication. Matched conics serve as guides to obtain preliminary estimates. When close approaches are calculated we simply use arcs of hyperbolas which are connected into heliocentric ellipses. But when the final complete orbit is to be established for actual operational purposes then the equations must be regularized so that accumulations of error at close approaches will be prevented. It is not inconceivable that the approximately 800 miles error occurring in the distance between the Mariner IV spacecraft and Mars at close approach, in spite of very high tracking accuracy, could have been significantly reduced by improved computational techniques.

Regularized integration programs not only give significantly increased accuracy but also reduced integration time. As our knowledge of the constants of the solar system increases, it becomes more and more reasonable to expect increased computational capability for lunar and interplanetary trajectories. In cases when the critical parts of the trajectory closely approach the singularities in the field, regularization is mandatroy in order to obtain meaningful results.

CONCLUSIONS

Regarding swing-by trajectories our present state of knowledge not only makes these acceptable but we may attach a certain natural expectation regarding their use. Presently existing or soon expected capability of propulsion and guidance, when combined with the idea of swing-by trajectories, will certainly open up new possibilities in space exploration.

Regarding capture-orbits, there is not much hope of finding free trajectories from the vicinity of the Earth such that the probe flying these trajectories becomes a permanent satellite of the Moon, Venus or Mars.

Both types of paths, the capture orbits and the swing-by trajectories, require regularization of the equations of motion in order to achieve sufficient computational accuracy and increased efficiency.

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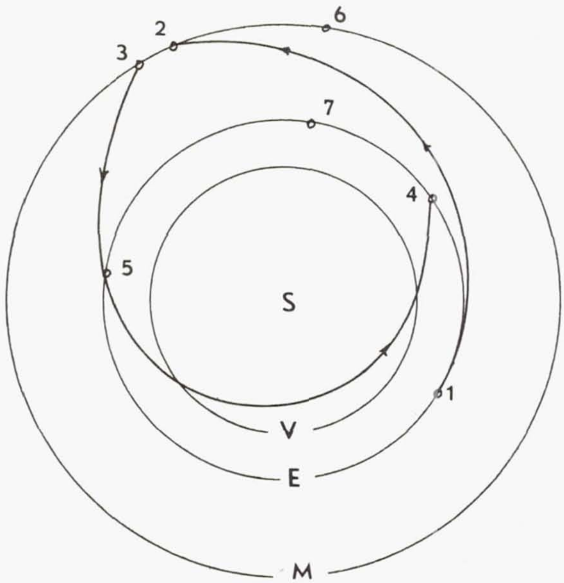

FIGURE 1

Standard round trip mission to Mars

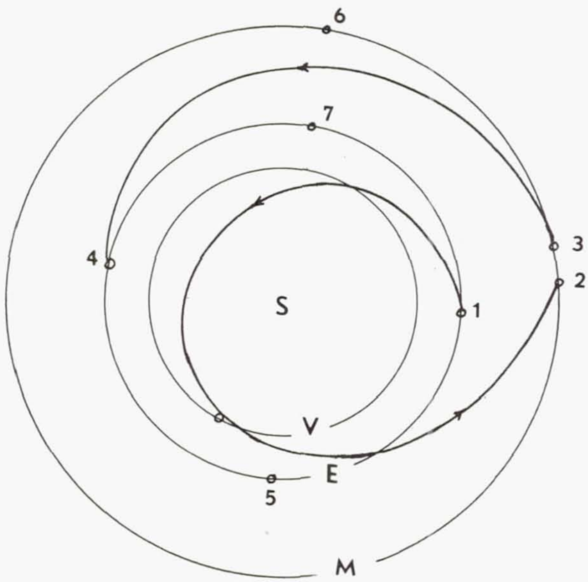

FIGURE 2

Round trip mission to Mars with Swing-by at Venus

TABLE I

Point	Description	Standard Trajectory	Swing-by Trajectory
1	Departure from Earth	V = 4.2 km/sec	V = 5.01 km/sec
1'	Swing-by Venus	--- ---	V = 9.9 km/sec 4 May 1979
2	Arrival at Mars	V = 6.9 km/sec 21 May 1980	V = 7.74 km/sec 4 September 1979
3	Departure from Mars	V = 6.45 km/sec 31 May 1980	V = 3.0 km/sec 14 September 1979
4	Arrival at Earth	V = 16.17 km/sec 16 January 1981	V = 4.44 km/sec 21 May 1980
5	Earth at time of arrival at Mars	1.1 A. U. from Mars	1.88 A. U. from Mars
6	Opposition (Mars)	25 February 1980	25 February 1980
7	Opposition (Earth)	25 February 1980	25 February 1980
-	Re-entry speed at Earth	V = 19.8 km/sec	V = 12.2 km/sec
-	Stop over at Mars	10 Days	10 Days
-	Trip-time from Earth to Mars (approx.)	6 Months	9 Months
-	Trip-time from Mars to Earth (approx.)	7.5 Months	8.3 Months
-	Total time of mission	430 Days	540 Days

TABLE 2

Planet	T_{sid}(mo)	T_{syn}(mo)
Mercury	2.9	3.8
Venus	7.4	19.2
Earth	12.0	∞
Mars	22.6	25.6
Jupiter	142.3	13.1
Saturn	353.5	12.4

TABLE 3

Planet	Mass-parameter	Egorov's result
Mercury	1.67×10^{-7}	no capture
Venus	2.45×10^{-6}	no capture
Earth	3.04×10^{-6}	no capture
Mars	3.23×10^{-7}	no capture
Jupiter	9.54×10^{-4}	?
Saturn	2.86×10^{-4}	?
Uranus	4.37×10^{-5}	?
Neptune	5.18×10^{-5}	?
Fluto	2.78×10^{-6}	no capture

It gives me exceptional pleasure to express my thanks to Drs. Cutting, Gates, Hornby and Ross of various NASA establishments who advised me regarding the present status of research on swing-by trajectories.

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SCIENTIFIC RESULTS OF MARINER MISSIONS TO MARS AND VENUS

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INTRODUCTION

Two successful spacecraft have been launched by the Mariner program. Mariner II reached the vicinity of Venus on December 14, 1962, and Mariner IV passed Mars on July 15, 1965. Both carried a carefully selected set of scientific instruments which were designed to answer current questions of primary importance about each planet. In 1962, the key question on Venus concerned the determination of the mechanism to explain the high temperature indicated by radio measurements. Thus, it was necessary to measure the brightness temperature of the planet at various wavelengths with sufficient spatial resolution to observe atmospheric limb darkening and positional variations. For Mars, in 1965, it was necessary to determine: 1) the structure of the atmosphere, and 2) the nature of the surface at higher resolution. For both planets, measurements were made of the exospheric structure in order to answer the questions whether or not Mars and Venus have Van Allen type radiation belts and hence a magnetic field. In this paper, I will discuss the methods used to answer these and other questions using the two spacecraft and their complement of sensors. The theme of the discussion will center on the critical questions of planetary exploration and how the Mariners have provided the early focus for this continuing study. Due to the obvious vastness of this subject, the approach here will be to note in outline form the essential aspects of this study providing references from current literature to fill in the details.

The separate disciplines in planetary sciences can be separated into five areas: 1) Planetary atmosphere, 2) Planetary interiors, 3) Planetary surfaces, 4) Planetary exospheres, and 5) Dynamical constants. Prior to the Mariner program, experimental data was limited to measurements of electromagnetic radiation emitted or reflected from the planet. This fact meant that many tortuous theoretical and experimental obstacles prevented the development of high confidence in the description of planetary conditions. Even with the recent attention to the near planets and the consequent development of better experimental tools such as high altitude balloons and large radio telescopes, many interesting fundamental questions remain essentially unanswerable. Selection of a payload, therefore, required application of several criteria in addition to their technical feasibility. First, it was clear that the relatively modest capability of the Mariner class of spacecraft meant that only experiments investigating the critical questions could be considered. Second, the experiment should require proximity to the planet and be impractical from less expensive terrestrial stations. Third, because each spacecraft would be the first to its respective planet, experiments should contribute to the collection of data necessary for the design of future missions. Finally, the well-defined, long-range goals of The National Aeronautics and Space Administration to search for and characterize extraterrestrial life and to understand the origin of the physical universe must be recalled to provide the comprehensive scientific rationale.

Before discussing specific results of the missions for each of the major disciplines of planetary science, I wish to describe each spacecraft and its complement of experiments. Literature giving more detailed discussions are readily available from the Jet Propulsion Laboratory.

The Mariner II spacecraft was developed in a very short period of time and was essentially a modest modification of the early Ranger design. It is shown in Figure 1. Three axis attitude stabilization provides a continuous stable orientation with the sun in the direction of the roll axis and the earth on the boresight of the

directional antenna. Communication to earth was normally through the directional antenna except during launch and midcourse maneuver when the spacecraft rotated away from its normal orientation. During these periods, the omni-antenna mounted on the superstructure was utilized. It also provided an emergency backup to a potential failure of the third axis stabilization.

In addition, the superstructure provided a remote mounting location required by three scientific instruments. Power was supplied by two solar panels whose individual dimensions were 60 inches by 30 inches. A trajectory correction maneuver to compensate for launch vehicle trajectory errors was accomplished by an axially mounted rocket motor (not shown). The experiments are all shown in position in the figure. Table 1 lists the experiments, a brief summary of their purposes and the responsible scientific investigators.

Mariner IV, while using the same basic subsystems as both the Ranger and Mariner II, incorporated innovations and improvements that were made possible by a longer development schedule. In addition, basic system approaches to the construction of spacecraft were improved to assure reliable operation over the long mission lifetime. Such things as exhaustive screening of electronic components and exhaustive testing at both the subsystem and system level resulted in increased confidence in the performance of the vehicle and in the validity of the scientific data. Figures 2 and 3 show two views of the spacecraft. Again, three axis stability is maintained, but now the star Canopus is used as the third axis reference replacing the earth. A fortunate happenstance of the Mariner IV trajectory was that the position of the earth in spacecraft coordinates was constant for the last 150 days of the mission. This allowed the use of a body-mounted, fixed-direction, high-gain antenna. The omni-antenna, mounted on the mast, was used for telemetry during launch, maneuvers and the first 100 days of the flight. Because of the larger transmitter power, uplink command capability was always possible through the omni-antenna. As for Mariner IV, the omni mast was used for mounting sensors which required remote locations. The experiments are shown in position by the two figures; and Table 2 lists the experiments, their purpose, and the responsible investigators.

In transit to Mars and Venus, many important new discoveries about the conditions in interplanetary space were made. The Mariner II solar plasma experiment measured a constant flux, low energy plasma boiling off of the Sun[1]. This discovery was confirmed by Mariner IV which further refined the energy spectra and direction of these particles[2]. The magnetometers on both spacecrafts characterized for the first time the detailed nature of the steady and disturbed interplanetary magnetic field[3,4]. The remaining particle detectors measured higher energy solar and galactic cosmic rays[5]. Both spacecraft carried microphones to detect the impact of micrometeteorites[6,7].

On both missions it was possible to select near-planet trajectory characteristics that satisfied the requirements for planetary observation without compromising the engineering constraints. On both missions trajectory corrections, made shortly after launch, placed the spacecraft on the desired course. Mariner II appraoched Venus along the trailing edge from outside the planet's orbit. Figures 4 and 5 illustrate the planetocentric geometry of the flight past Venus. At about 65 minutes before closest approach, or at a distance of about 47,400 km from the planet's center, the radiometer began to scan the planet. At a distance of about 41,800 km from the planet because of the angular movement of the spacecraft in its hyperbolic orbit about Venus. Table 3 gives the Aphrodiocentric orbital elements of the Mariner II trajectory.

The Mariner IV passage of Mars had to be chosen to permit Earth occultation, good Television coverage, and penetration of the potential Martian magnetosphere, without violation of engineering and operational constraints. The final aiming point was nearly optimum from all of these standpoints. Figure 6 shows the near-Mars trajectory. The Sun is approximately in the direction perpendicular to the R-T plane, and the north pole of Mars is approximately in the direction of the R vector. Table 4 gives the aerocentric orbital elements of the Mariner IV trajectory.

DYNAMICAL CONSTANTS

Precise analysis of the trajectory past the planets, especially if the bending is large, will improve the accuracy of the values of the astronomical unit, the planetary and satellite mass, and the planetary ephemerides. Both Mariners made improvements in the knowledge of these parameters, and it should be noted that incorporation of a ranging measurement on such missions would be of further benefit. The analysis of the Mariner II data is given in references 8 and 9. At this time the Mariner IV results are unpublished.

Radar measurements of the astronomical unit have given values ranging from 149, 597, 900 to 149, 599, 800. Mariner II gave a value of 149, 599, 400 with somewhat smaller error. The Mariner IV result is at present ambiguous because there was an unexplained discrepancy in the position of Mars that appeared suddenly in the operational trajectory run just prior to encounter. This anomaly is unexplained and according to the trajectory analysts it may remain unexplained for some time. Mariner IV trajectories were computed using the Mariner II value for the astronomical unit.

The two missions gave improved values for the masses of Venus, Mars, and the Moon. From Mariner II the ratio of the Earth-Moon masses is 81.30155 compared to less accurate values from Rangers VI and VII of 81.3036 and 81.3044. The uncertainty is 0.001. The ratio of the mass of the Sun to that of Venus from Mariner II is 408,607 with the uncertainty in the next to last digit. Figure 7 shows the history of the determination of the mass of Mars. The Mariner IV value of 3,098,600 for the ratio of the Sun's mass of that of Mars has an uncertainty of 3,000.

PLANETARY EXOSPHERES

Both Mariners carried a sizeable complement of experiments designed to characterize the particle and field environments of Mars and Venus. The discovery of the Van Allen belts surrounding the Earth was the first major discovery using the newly-acquired satellite capability. Many Earth satellites beginning with Explorer I had measured the intensity and distribution of particles trapped in the planetary magnetic field. This data had been formulated into a relatively secure theory based on solar and cosmic ray partical injection, magnetic field trapping, and atmospheric extraction. A magnetometer on Lunik I had confirmed the suspicion that the moon had a very small field; and, consequently, no particle belts. Terrestrial measurements of Zeeman splitting and synchrotron radiation had given measurements of magnetic fields on the Sun and for Jupiter. Prior to Mariner II the only experimental indication of a Venusian field was by Houtgast[10] who correlated the terrestrial magnetic index with the time of Venus inferior conjunction. His results, being based on the hypothesis that Venus would deflect solar particles, were admittedly uncertain and gave a value for the Venusian field of five times that of the Earth. Theoretically, the Venusian field was judged to be small based on its presumed slow rotation rate of 225 days. All instruments on Mariner II (Magnetometer, Geiger tubes, Plasma, and Ion Chamber) gave a null result. A series of papers by the appropriate experimenters [11,12,13] analyzed this result in terms of the trajectory and the theory of the interaction of the solar plasma with a planetary field. Their collective conclusion put the experimental upper limit of the Venusian magnetic moment at 0.18 that of Earth. Later analysis by Van Allen[14] lowered this value to about 0.10.

Before the Mariner IV encounter, the only experimental data on the Martian field was one uncertain radio measurement by Davies[15] which indicated synchrotron radiation from a very intense (10^6 times the Earth) trapped particle belt. All theoretical predictions based on the 24 hour, 37 minute rotation rate and a reasonable internal structure indicated the Martian pole strength was, at most, one-tenth that of the Earth. It was fortunate that the intense belts were not present, because it was extremely doubtful that the Mariner IV spacecraft could operate successfully in such an environment. The actual results from Mariner IV are presented in a series of articles by each experimenter (Magnetometer, Trapped Radiation Detector, Cosmic Ray Telescope, Plasma) in Science[16]. Because the trajectory passage of Mars was more favorable for detecting the magnetopause than that by Venus, it was possible to lower the upper limit of the Martian dipole strength to 1/300 that of Earth.

The theory of planetary exospheres seems to be established on an increasingly firm theoretical foundation. The Mariners have been the primary experimental tools in this study. Both Mars and Venus apparently provide interesting instances of the solar plasma inpinging directly on the planetary atmosphere.

PLANETARY ATMOSPHERES

The structure and composition of the atmospheres of Mars and Venus had been the object of intensive investigation prior to the Mariner flights. In retrospect, it appears that some of the observations in which there was high confidence were disproved; and, conversly, some of the suspect data was confirmed. In 1962, there were three competing theories attempting to explain the apparent high surface temperature of Venus indicated by the radiometric measurements: 1) the "greenhouse"

theory, 2) the "dust-bowl" theory, and 3) the "ionosphere" theory. None of these could satisfactorily account for all the data, a situation which still exists.

The microwave radiometer and the infrared radiometer experiments took advantage of the passage near Venus to obtain high spatial resolution temperature measurements. Earth-based measurements of the radio emission of Venus had indicated that the planet's temperature was approximately 600° K for wavelengths in excess of 3 cm. This temperature may be contrasted with infrared measurements of Venus which yield values somewhat less than half those obtained by radio telescopes. The radio data, which are critical to our understanding of the Venusian environment, rest on terrestrial observations which suffer from lack of spatial resolution and insufficient precision. Flyby planetary probes offer the possibility of precision and resolution with modest radiometers. Accordingly, the Mariner II spacecraft was instrumented with a two-channel microwave radiometer operating at wavelengths of 13.5 and 19.0 mm.

The pertinent equipment performance parameters are given in Table 5. The effective antenna gain was calibrated by using a black disk of known temperature, whose angular size was designed to be approximately the size of Venus at encounter.

During the 109-day flight, 23 noise calibrations were made. Thus, the gain, base-level, and time-constant performance of the radiometers could be monitored en route.

The radiometers were energized, and the antenna scan motion was activated about $6\frac{1}{2}$ hours before encounter. The scan motion had an angular extent of 123.5° and a nominal scan rate of 0.1 deg/sec. The microwave radiometer first made contact with the planet Venus at 18:59 GMT (spacecraft time) on December 14, 1962. During the next 35 min, three scans across the planetary disk were obtained, as follows (see Figure 8).

Scan	Approx. angular extent, deg	Alt. at mid-scan, km	Location
1	10	40 200	Dark side.
2	15	37 750	Near terminator.
3	10	35 850	Light side.

Telemetered digital data points constituted the basic data, which had to be corrected for a number of effects before they could be considered as yielding the microwave temperature distribution across the planet. Among these corrections were the more important effects of the post-detection time constant and a detailed consideration of the antenna pattern.

The noise tube calibrations obtained en route to Venus made it possible to determine the in-flight time constant and gain of the radiometers. The gain of both channels decreased during the cruise, and the zero levels had systematic variations. These effects were more serious in the 13.5-mm radiometer.

Preliminary estimates of the peak-brightness temperatures of the three scans were: Scan 1 (dark side), 460° K; scan 2 (near terminator), 570° K; scan 3 (light side), 400° K. The temperatures are based on calculations which account for the effects of the antenna beam and the postdetection time constant. The errors of the quoted temperatures are estimated to be 15%. The analysis of the preliminary results suggests that there is no significant difference in the microwave temperatures on the light and dark sides of the planet. The results suggest a limb-darkening, an effect which represents cooler temperatures near the edge of the planetary disk. The ionosphere model of the Venus atmosphere, which permits Earth-like temperatures, appears to be ruled out by these observations. On the other hand, the observed limb-darkening is consistent with a model of the Venusian environment which has high temperatures originating deep in the atmosphere or at the surface of the planet.

Thus, Mariner II found an unquestionable limb-darkening and also found that there is little difference in temperature on the dark side compared with the sunlit side of the planet. On the basis of the radiometer scans, the surface of Venus, where the 19-mm radiation originates, appears to have a temperature of about 400° K.

The infrared radiometer which was flown on Mariner II in conjunction with the microwave radiometer was designed to measure, with high geographical resolution, the infrared radiation from Venus in two wavelength regions. One of these was centered

on the 10.4μ carbon dioxide band, while the other was selected to correspond to an infrared window centered at 8.4μ. The infrared radiometer was mounted upon and boresighted with the microwave radiometer. Both instruments, therefore, executed the same scan pattern caused by the combined effects of the probe motion and a rotation of the radiometers in a plane normal to the probe-Sun line. From the three scans of the planet, five pairs of radiation temperatures were obtained on the dark side, five on the sunlit side, and eight along the terminator.

The data are consistent with an equality of the 8μ and 10μ radiation temperatures. This apparent equality would indicate that there was little carbon dioxide absorption in the light path. The implications are that the measured temperatures were cloud temperatures, that the clouds were quite thick, and that essentially no radiation was transmitted from the surface.

A definite limb-darkening was observed in both spectral channels; the radiation temperatures showed a monotonic decrease of approximately 20° K between the central region and the limbs. Central radiation temperatures are estimated to be on the order of 240° K. The data do not show any clear-cut evidence of asymetry in the limb-darkening, except for an anomaly on the southern part of the terminator scan. In particular, the light- and dark-side temperatures were qualitatively the same. The anomaly was about 10° K cooler than expected on the basis of symmetrical limb-darkening. One obvious interpretation of this temperature anomaly is that the clouds were locally higher, or more opaque, or both.

The improvement of knowledge of the atmosphere and ionosphere of Mars has long been an important scientific objective of astronomers and other investigators. Recently, the technological value of this knowledge has been greatly enhanced by the need to obtain more accurate information on the physical properties of the Martian atmosphere that are needed for the design of survivable landing capsules to perform perhaps the most important experiments of planetary exploration - those in search of extraterrestrial life.

The present knowledge of such atmospheric properties as the surface pressure and scale height is quite inexact. The surface pressure, as deduced from recent spectroscopic observations[17], was thought to lie between 10 and 25 millibars, in contrast to the 85 millibar figure previously derived from Rayleigh scattering measurements. The vertical structure of the atmosphere, including the properties of the troposphere and the scale height in the stratosphere, are not accessible to direct earth-based measurement, and can, therefore, only be estimated on the basis of assumptions of atmospheric constituents and temperatures. Likewise, the properties of the Martian ionosphere have been open only to postulation of models based in turn on the estimated structure of the Martian upper atmosphere. Current models indicate that the peak electron density might be between 10^{11} and 2×10^{13} el./m^3 (reference 4).

The preliminary results of the occultation experiment will be reported soon[18]. Estimates of the refractivity and density of the atmosphere near the surface, the scale height in the atmosphere, and the electron density profile of the Martian ionosphere have been obtained. The atmospheric density, temperature, and scale height are lower than previously predicted, as are the maximum density, temperature, scale height, and altitude of the ionosphere.

The geometry of the occultation is given in Figure 4. At entry into occultation, the spacecraft was at a distance of 25,570 km. from the limb of Mars, traveling at a velocity of 2.07 km/sec normal to the Earth-Mars line. The point of tangency on the surface of Mars was at a latitude of 55° S and a longitude of 177° E. At the time of exit from occultation, the distance from the limb of Mars had increased to 39,130 km, and the point of tangency was located at about 60° N and 34°W.

In brief, the results appear to be as follows. The ionosphere peaks between 120 and 150 km above the surface and has a maximum density of 10^5 electrons/cc. Best fit to the data is given with a heavy atomic weight suggesting the preponderance of CO_2 in the atmosphere. The surface refractivity is between 3.7 and 4.2, and the scale height is between 8 and 9 km. A nominal surface pressure of 5 mb. is indicated with the obvious reservation that this corresponds to the pressure at the point of tangency which is probably somewhat higher than the average surface.

PLANETARY SURFACES

At this time the results of the Mariner television experiment are summarized in the August 6, 1965 [19] issue of Science, which I will just reproduce in its entirety, concluding by showing several of the pictures.*

*Reprinted by premission of the American Association for the Advancement of Science.

The Mariner IV spacecraft successfully acquired and transmitted to earth 22 picutres of the planet Mars taken from a distance of 17,000 to 12,000 km just before its closest approach to Mars at approximately 00:30 UT on 15 July 1965. This first report describes the performance of the television camera system, the resultant picture quality, and the more prominent surface features present in the picture. We feel that some of these features are so striking that certain physical and geological inferences can be drawn even at this early date.

The complete set of pictures in their current state of processing was released to the scientific community and the public on 29 July 1965. The completely processed pictures, the relevant calibration data, and a more detailed analysis of surface features will be presented later.

In regard to the design and performance of the TV system, one of the most difficult problems associated with the Mariner photographic mission to Mars was the wide illumination range and the low surface contrast to be expected. Since the camera would be viewing the surface from the bright limb to, and beyond, the evening terminator, the camera was called upon to respond to brightness ranging from full solar illumination near the sub-solar point to near total darkness as the terminator was crossed. The slow-scan vidicon chosen was capable of handling a 30-to-1 range of illumination with fixed operating voltages. The low communications rate from the planet required a digital transmission system. In order to effectively utilize a high signal-to-noise ratio the video signal was divided into 64 equal increments. Since the video signal level would decrease as the photo path approached the terminator, automatic video gain control was incorporated. The control was designed to maintain a video level which would contain at least 15 of 64 increments. The camera telescope was of the Cassegrain type with 12-inch (30-cm) focal length, f/8 focal ratio, and 0.2-second shutter time.

The camera operated at the minimum gain until picture No. 18, in which the video level fell below the minimum allowed. The system then increased gain for picture No. 19 which was taken 96 seconds later. As the terminator was approached, the gain increased a second time for picture No. 20 and reached its maximum for pictures No. 21 and No. 22. The automatic adjustment of video signal level was not able to fully cope with the unexpectedly low light intensity, and pictures No. 15 through 20 show decreasing signal-to-noise ratios. In addition, a spurious background brightness was detected on picture No. 1 apparently at a distance of more than 100 km from the limb of the planet. Preliminary analysis has indicated this spurious background to be approximately one-fourth the brightness of the planet itself. Similar analysis of pictures Nos. 21 and 22, which were taken on the dark side of the terminator, reveals considerably lower levels of brightness, about 1/8th and 1/25th that of picture No. 1, respectively. This background is tentatively attributed to an instrumental defect of an optical nature which developed during the $7^1/_2$-month space flight. All other camera characteristics such as resolution and geometrical fidelity appear normal. Results of the first tape playback have indicated that the tape machine and communications equipment operated as designed.

In pictures No. 1 through No. 4, the very high solar illumination of the terrain viewed by the camera significantly reduced the visibility of surface features, as had been anticipated. Pictures No. 5 through No. 14, however, present a view of a densely cratered surface, closely comparable to bright upland areas of the Moon

We have observed more than 70 clearly distinguishable craters ranging in diameter from 4 to 120 km. It seems likely that smaller craters exist; there also may be still larger craters than those photographed, since Mariner IV photographed, in all, only about 1 percent of the Martian surface.

The observed craters have rims rising to about 100 m above the surrounding surface and depths of many hundred meters below the rims. Crater walls so far measured seem to slope at angles up to about 10°. The number of large craters present per unit area on the Martian surface and the size distribution of those craters resemble remarkably closely the lunar uplands, as illustrated in Figure 9 and 10 and reference 20.

If the Mariner sample is representative of the Martian surface, the total number of craters of the sizes so far observed is more than 10,000 compared to a mere handful on Earth. In appearance, the Martian craters closely resemble impact craters on Earth, both artificial and natural, and the craters of the Moon. Craters of widely different degree of preservation and, presumably, age are distinguished. A few elongate markings of diffuse nature are present on the Mariner photos but at this early stage of analysis no conclusions can be offered concerning them. On frame No. 13, one such feature looks like a part of the edge of a very large crater and, perhaps significantly, lies near the border of a Martian dark area. In southern subpolar latitudes, where the season is now late midwinter, some craters appear to be rimmed with frost, particularly those in frame No. 14.

Some mention must be made of features looked for, but not seen, on the Mariner photos. Although the line of flight crossed several "canals" sketched from time to time on maps of Mars, no trace of these features was discernible. It should be remembered in this respect that the visibility of many Martian surface features, including the "canals," is variable with time. No Earth-like features, such as mountain chains, great valleys, ocean basins, or continental plates were reorganized. Clouds were not identified, and the flight path did not cross either polar cap.

Although it may be difficult to ever arrive at an unambiguous identification and interpretation of all the features recorded on the Mariner photographs, we feel that the existence of a lunar-type cratered surface, even in only a 1-percent sample, has profound implications about the origin and evolution of Mars and further enhances the uniqueness of Earth within the solar system. By analogy with the Moon, much of the heavily cratered surface of Mars must be very ancient--perhaps 2 to 5×10^9 years old[21]. The remarkable state of preservation of such an ancient surface leads us to the inference that no atmosphere significantly denser than the present very thin one has characterized the planet since that surface was formed. Similarly, it is difficult to believe that free water in quantities sufficient to form streams or to fill oceans could have existed anywhere on Mars since that time. The presence of such amounts of water (and consequent atmosphere) would have caused severe erosion over the entire surface.

The principal topographic features of Mars in the areas photographed by Mariner have not been produced by stress and deformation originating within the planet, in distinction to the case of Earth. Earth, of course, is internally dynamic, giving rise to mountains, continents, and other such features, whereas Mars has evidently long been inactive. The lack of internal acitvity is also consistent with the absence of a significant magnetic field on Mars, as determined by the Mariner magnetometer experiment.

As we had anticipated, Mariner photos neither demonstrate nor preclude the possible existence of life on Mars. Terrestrial geological experience would suggest that the search for a fossil record appears less promising if Martian oceans never existed. On the other hand, if the Martian surface if truly "near pristine," that surface may prove to be the best--perhaps the only--place in the solar system still preserving clues to primitive organic development, traces of which have long since disappeared from Earth.

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TABLE 1

Mariner Scientific Experiments - Venus 1962

Instrument	Purpose	Experimenter
Microwave	Determine the Temperature of the Planet Surface and Details Concerning its Atmosphere	A. H. Barrett - MIT D. E. Jones - JPL J. Kopeland - AOMC A. E. Lilley - HARVARD
Infrared Radiometer	Determine Any Fine Structure of the Cloud Layer	L. D. Kaplan- JPL G. Neugebauer - JPL C. Sagan - UCB
Magnetometer	Measure Changes in the Planetary and Interplanetary Magnetic Fields	P. J. Coleman - NASA L. Davis - CIT E. J. Smith - JPL C. P. Sonnett - NASA
Ion Chamber and Particle Flux Detector	Measure Charged Particle Intensity and Distribution in interplanetary Space and in the Vicinity of the Planet	H. R. Anderson - JPL H. V. Neher - CIT J. Van Allen - SUI
Micrometeorite Detector	Measure the Distribution of Micro-meteorites	W. M. Alexander - GSFC
Solar Plasma Spectrometer	Measure the Intensity of Low Energy Protons from the Sun	M. Neugebauer - JPL C. W. Snyder - JPL

TABLE 2

Mariner Scientific Experiments - Mars 1964

Experiment	Purpose	Principal Investigator
Television Subsystem	Obtain Close-Up Pictures of the Planet Surface	R. Leighton - CIT
Occultation	Obtain Data Relating to Scale Height and Pressure in the Atmosphere of Mars	A. Kliore - JPL
Magnetometer	Measure Magnitude and Other Characteristics of the Planetary and Interplanetary Magnetic Fields	E. Smith - JPL
Ion Chamber	Measure Charged Particle Intensity and Distribution in Interplanetary Space and in the Vicinity of the Planet	H. Neher - CIT
Trapped Radiation Detector	Measure Intensity and Direction of Low-Energy Particles	J. Van Allen - SUI
Cosmic Ray Telescope	Measure Direction and Energy Spectrum of Protons and Alpha Particles	J. Simpson - U. of CHI.
Cosmic Dust Detector	Measure Momentum Distribution, Density, and Direction of Cosmic Dust	W. Alexander - GSFC
Plasma Probe	Measure the Very Low-Energy Charges Particle Flux from the Sun	H. Bridge - MIT

TABLE 3

Aphrodiocentric Orbital Elements of Mariner II Trajectory

Hyperbolic Orbital Element	Venus-Encounter Orbit
Semimajor axis, a , km	10,971.61
Eccentricity, e	4.732 749
Inclination to ecliptic, i , deg	134.899 3
Longitude of ascending node, Ω , deg	216.745 8
Argument of periapsis, ω , deg	236.826 7
Periapsis distance, q , km	40,954.24
Time of periapsis passage, T , GMT	Dec. 14, 1962

TABLE 4

Aerocentric Orbital Elements of Mariner IV Trajectory

Hyperbolic Orbital Element	Mars-Encounter Orbit
Semimajor axis, a , km	-2,209.21
Eccentricity, e	6.97526
Inclination to ecliptic, i , deg	58.1858
Longitude of ascending node, , deg	187.4988
Argument of periapsis, ω , deg	289.3206
Periapsis distance, q , km	13,200.6
Time of periapsis passage, T , GMT	01:00:58.12 July 15, 1965

TABLE 5

Microwave radiometer characteristics

Parameter	Channel 1	Channel 2
Center wavelength, mm	19	13.5
Center frequency, Gc/sec	15.8	22.2
Predetection bandwidth, Gc/sec	1.5	2.0
Sensitivity, rms, °K	15	15
Calibration signals, °K	1500	800
Time constant, sec	40	40
Beamwidth, deg	2.5	2.2
Side lobes, db	-23	-23
Reference frequency, cps	950	1050

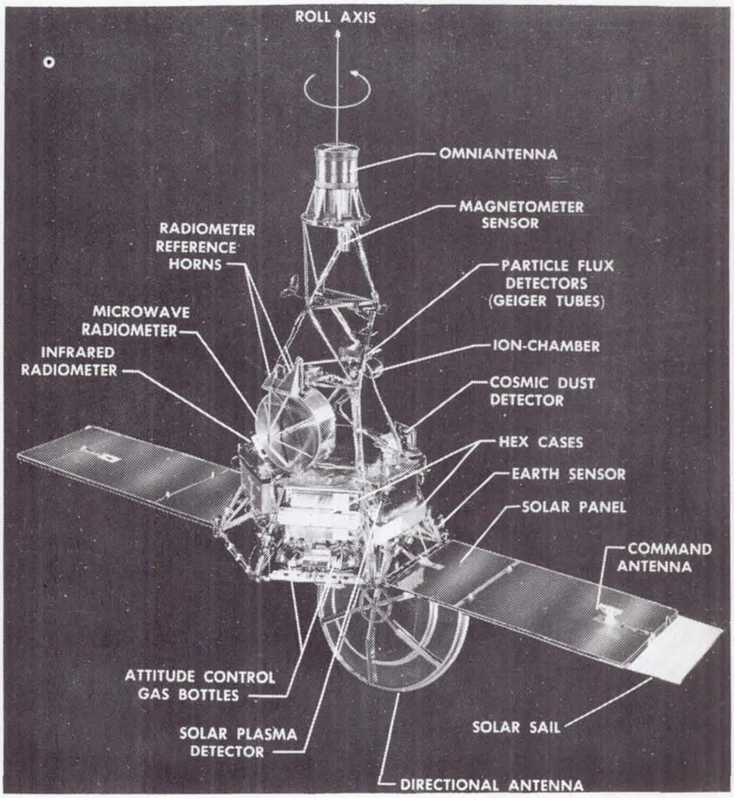

FIGURE 1

Mariner II Spacecraft

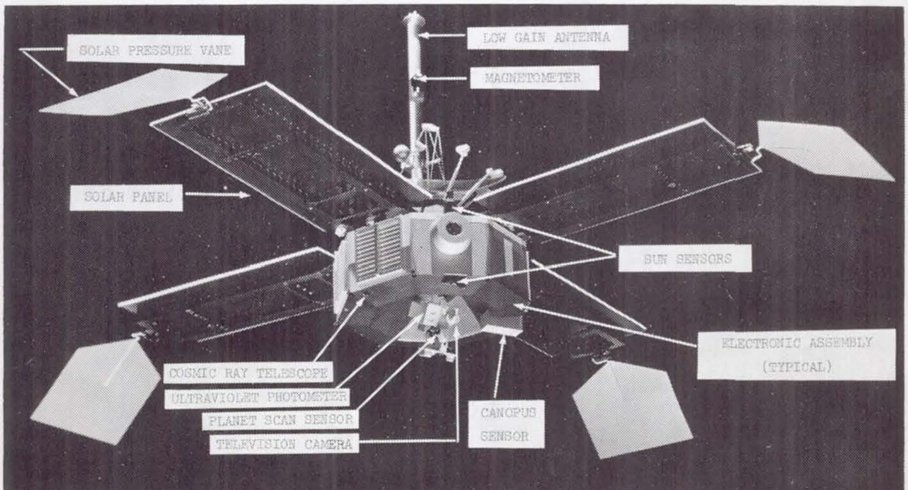

FIGURE 2

Mariner IV Spacecraft

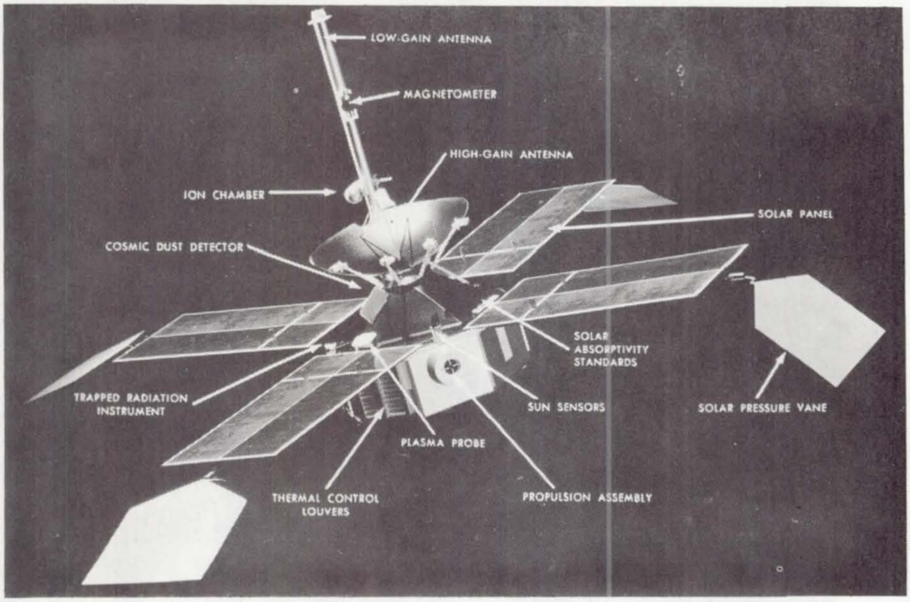

FIGURE 3

Mariner IV Spacecraft

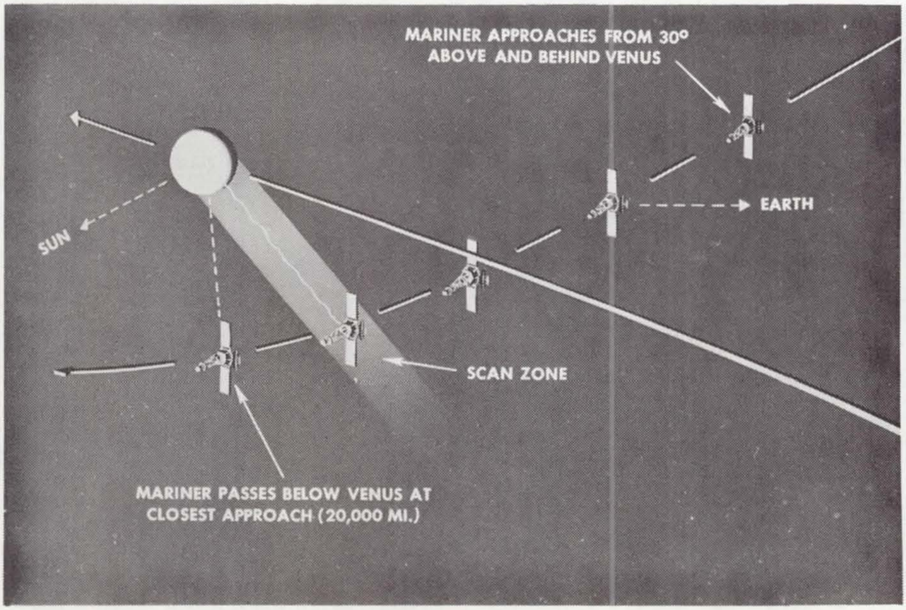

FIGURE 4

Mariner II Pass of Venus as seen from Inside Venus Orbit

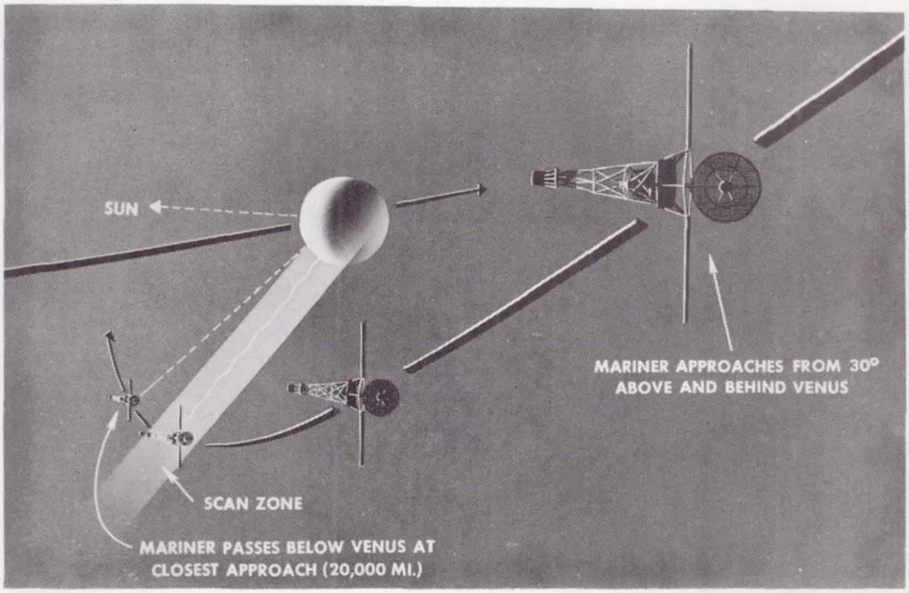

FIGURE 5

Mariner II pass of Venus as seen from Earth

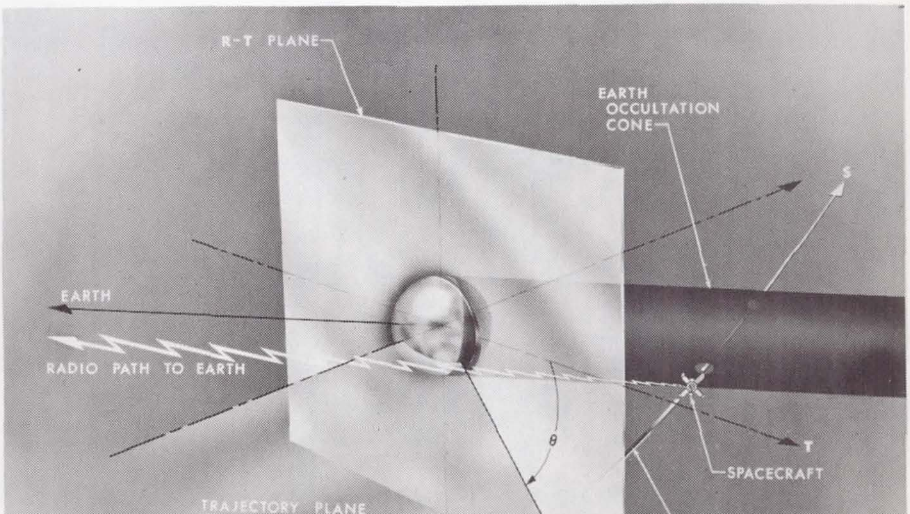

FIGURE 6

Near-Mars Trajectory

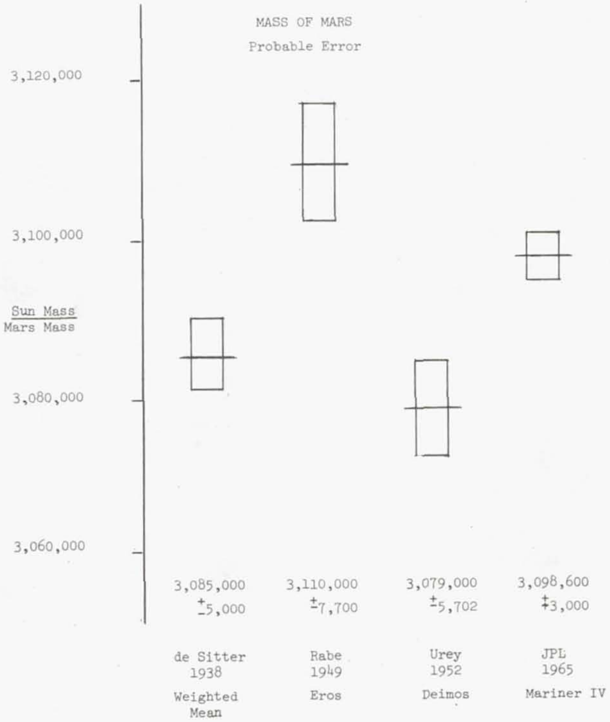

FIGURE 7

History of the Determination of the mass of Mars

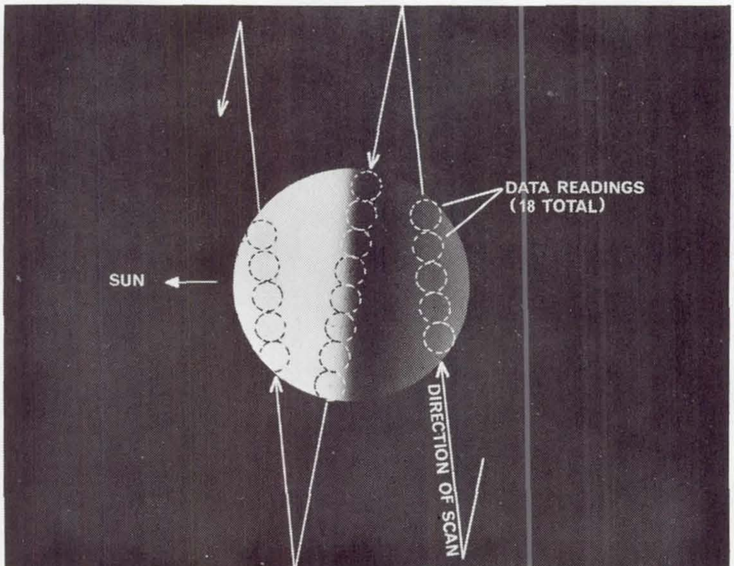

FIGURE 8

Mariner II Radiometer Scans of Venus

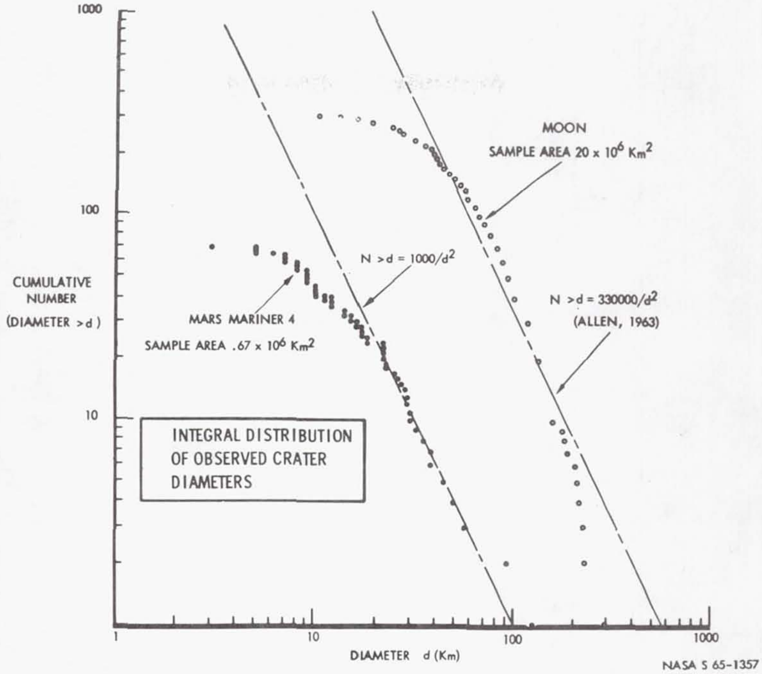

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FIGURE 9

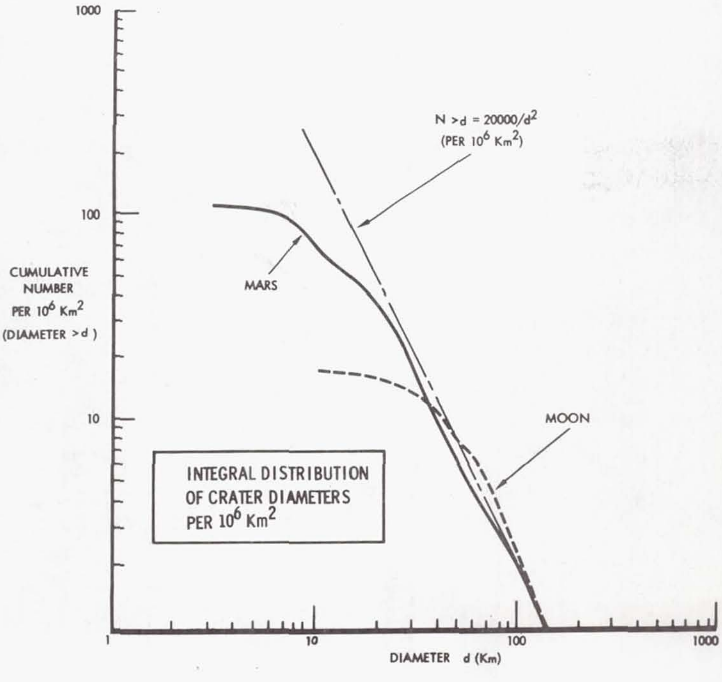
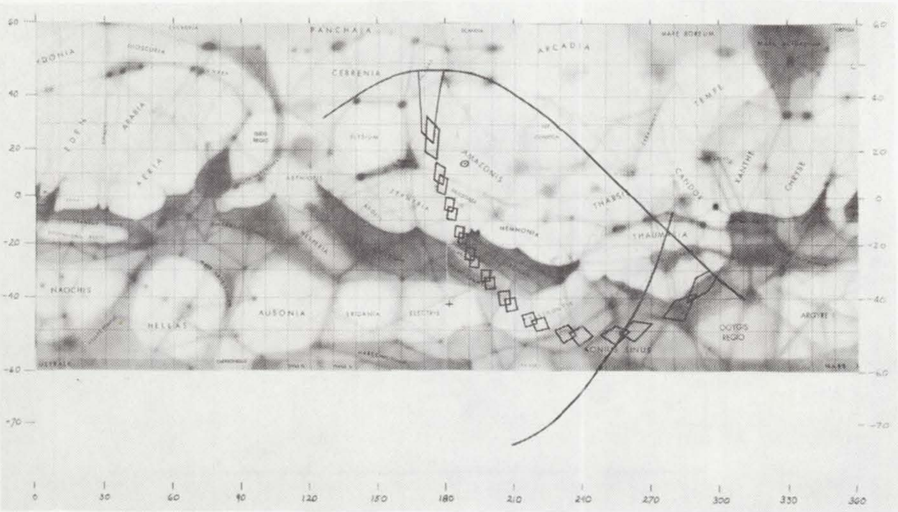
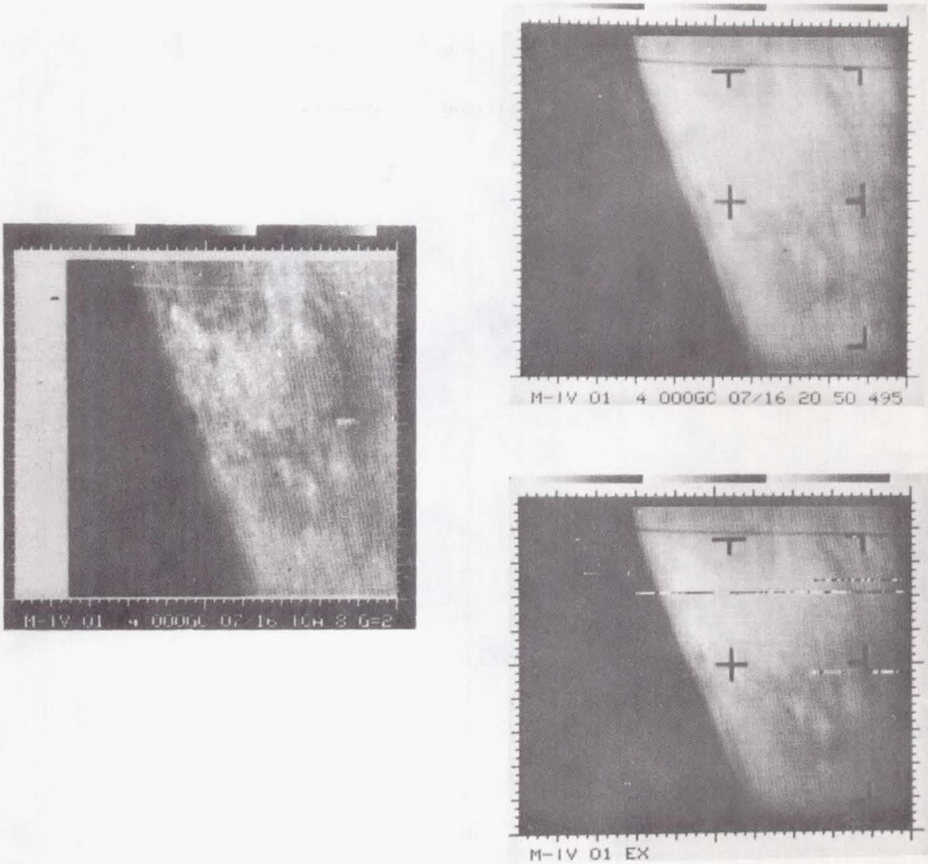


FIGURE 10



This Mars map shows the area on the planet's surface covered in 21 pictures and a fraction of a 22nd picture taken by Mariner IV's television camera on July 14, 1965, and recorded on tape for playback to space communications stations on Earth. Total area photographed was about 600,000 square miles, approximately one per cent of the entire Martian surface. Mariner's camera scanned from north to south, recording pairs of overlapping pictures in a red-green, green-red, filter sequence. The first picture, which captured the limb of Mars against the background of space, was taken from a slant range of 10,500 miles. Closest distance between the camera and the area photographed was 7,400 miles. Mariner IV was launched November 28, 1964, from Cape Kennedy. It flew by Mars on the 228th day of the mission at a closest approach distance of 6118 miles. Picture playback began the following day--July 15, 1965--and was completed July 24. Second playback of the pictures ended August 2, the 247th day of flight, when Mariner's telemetry system was returned to cruise mode to obtain additional fields-and particles measurements and spacecraft engineering information. Planetary science data, including the TV pictures, were transmitted to Earth over distances ranging from 134 million to 150 million miles.



MARINER IV PICTURE NO. 1

Viewed with data block at left, North is at top. Sun is 25° from the zenith, from the southeast in the photo.

Time Taken: 5:18:33 p.m. PDT, July 14, 1965.

Slant Range: 10,500 miles.

Area Covered: Along the limb: about 410 miles. From limb to edge of the photo: about 800 miles.

Location (picture center): 35° North Latitude, 172° East Longitude.

Map Description: Bright region between Trivium Charontis and Propontus II Phlegra, a bright region, is on the limb.

Filter: Orange.

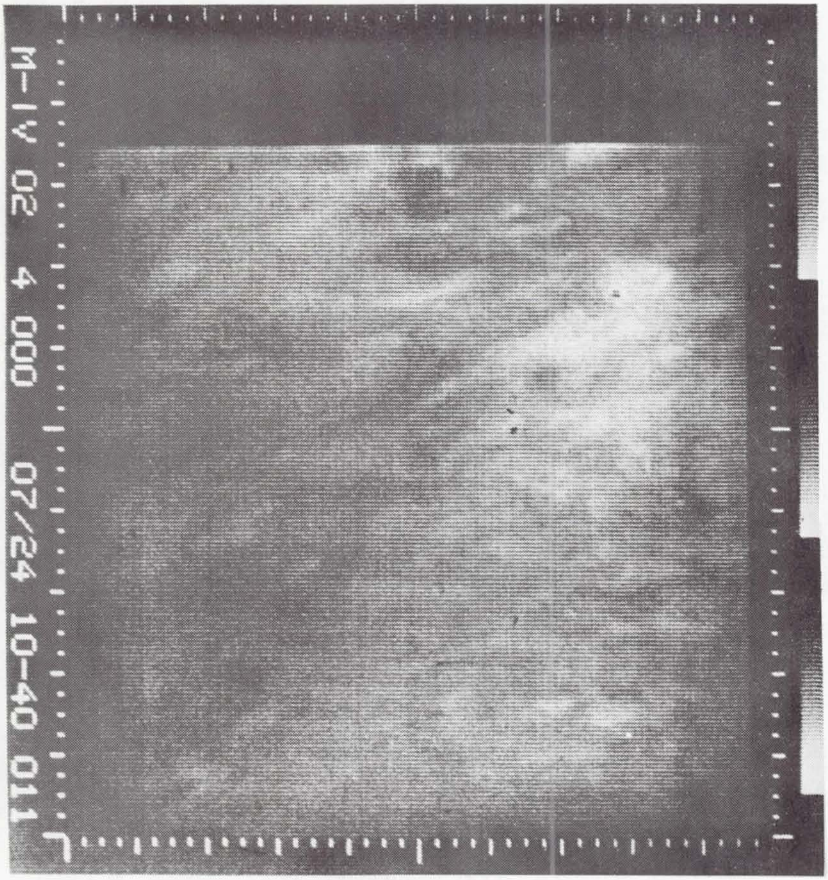
Overlap: Lower right corner overlaps picture number 2.

REMARKS:

Top Frame: Most recent intermediate step of data processing, including contrast enhancement factor of two and fiducial marks removed.

Lower Left Frame: Raw picture.

Lower Right Frame: With preliminary processing as released Thursday, July 15, 1965.



MARINER IV PICTURE NO. 2

Viewed with data block at left, North is at top. Sun is 20° from the zenith, from the southeast in the photo.

Time Taken: 5:19:21 p.m., PDT, July 14, 1965.

Slant Range: 10,100 miles.

Area Covered: East-West: 290 miles. North-South: 530 miles.

Location: 27° North Latitude, 174° East Longitude.

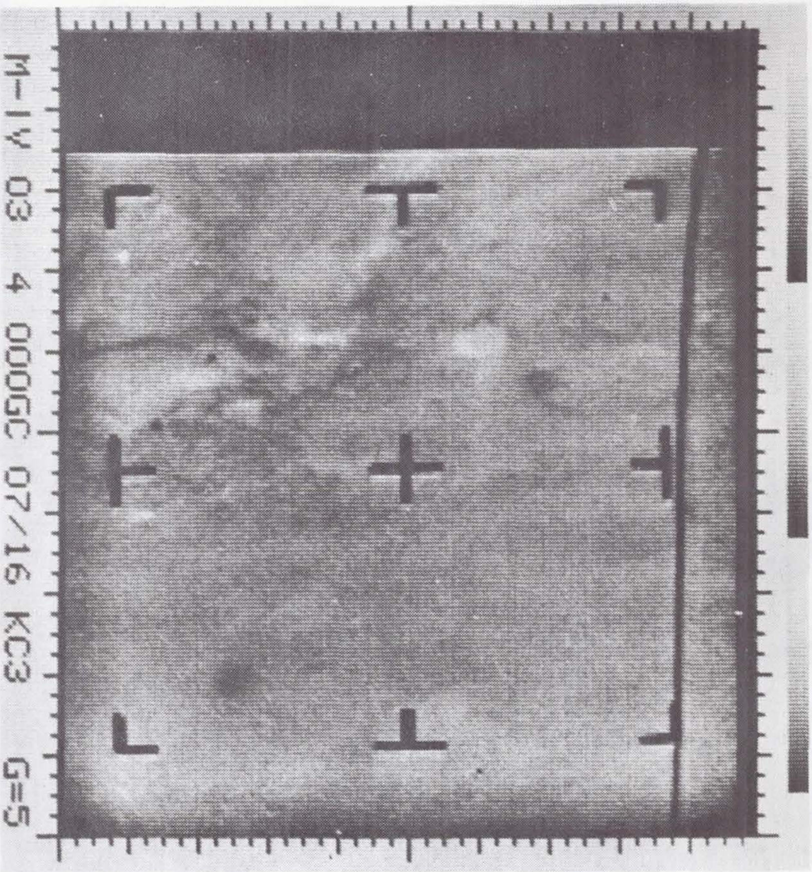
Map Description: Bright region northeast of Trivium Charontis.

Filter: Green.

Overlap: Upper left corner overlaps picture number 1.

REMARKS:

The picture shown has a contrast enhancement factor of two.



MARINER IV PICTURE NO. 3

Viewed with data block at left, North is at top. Sun is 14° from the zenith, from the East in the photo.

Time Taken: 5:20:57 p.m., PDT, July 14, 1965.

Slant Range: 9500 miles.

Area Covered: East-West: 200 miles. North-South: 310 miles.

Location: 13° North Latitude, 177° East Longitude.

Map Description: Bright region southeast of Trivium Charontis.

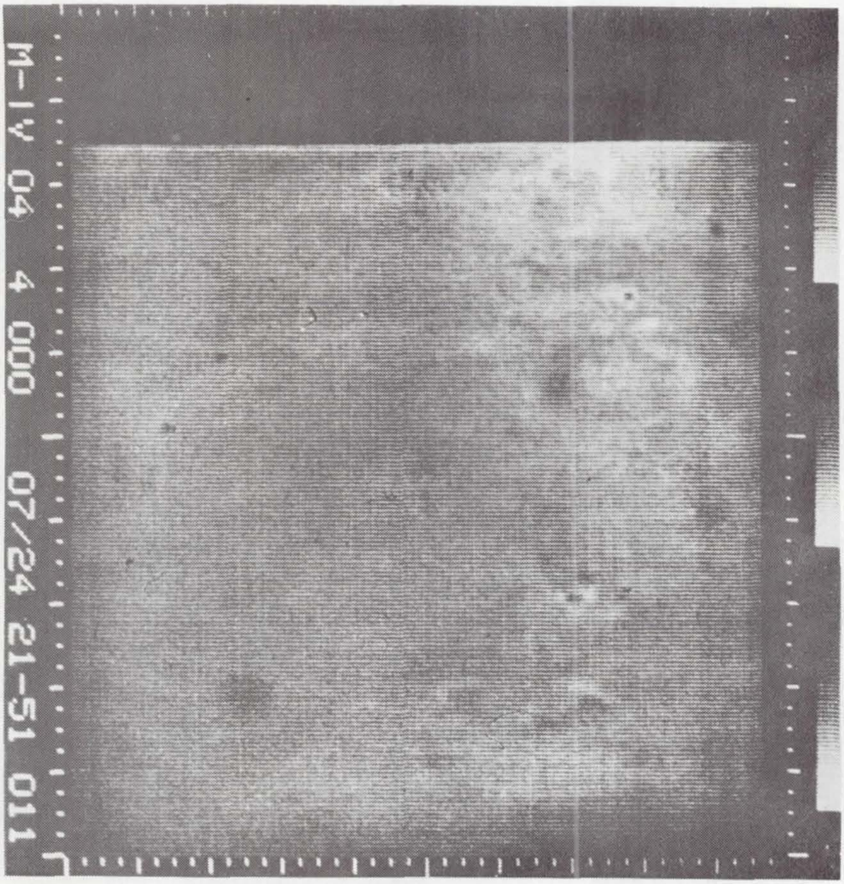
Filter: Green

Overlap: Lower right corner overlaps picture number 4.

Contrast Enhancement Factor: Five

REMARKS:

Considerable fine tonal detail is apparent, but differentiation between topographic features and surface reflectivity variations is particularly difficult under the lighting and viewing conditions under which this photograph was taken.



MARINER IV PICTURE NO. 4

Viewed with data block at left, North is at top. Sun is 14° from the zenith, from the northeast in the photo.

Time Taken: 5:21:45 p.m., PDT, July 14, 1965.

Slant Range: 9300 miles.

Area Covered: East-West: 210 miles. NOrth-South: 270 miles.

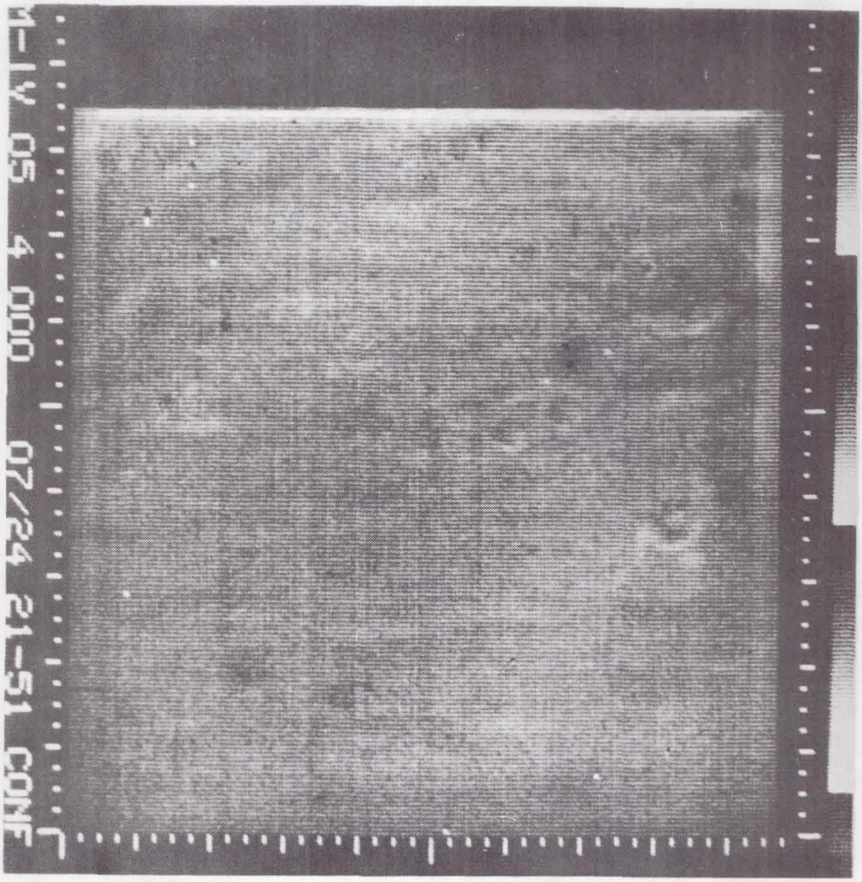
Location: 7° North Latitude. 179° East Longitude.

Map Description: Bright region in Mesogaea.

Filter: Orange.

Overlap: Upper left corner overlaps picture number 3.

Contrast Enhancement Factor: Two.



MARINER IV PICTURE NO. 5

Viewed with data block at left, North is at top. Sun is 19° from the zenith, from the North in the photo.

Time Taken: 5:23:21 p.m., PDT, July 14, 1965.

Slant Range: 8900 miles.

Area Covered: East-West: 190 miles. North-South: 220 miles.

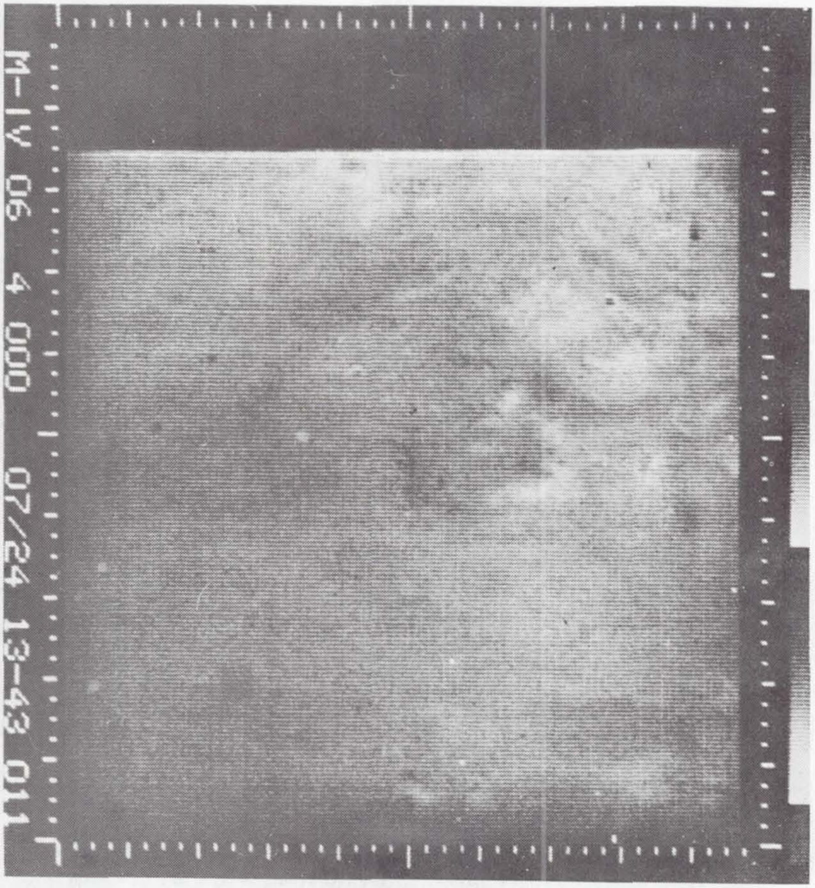
Location: 2° South Latitude, 181° East Longitude.

Map Description: Bright region in eastern Zephyria.

Filter: Orange.

Overlap: Lower right corner overlaps picture number 6.

Contrast Enhancement Factor: Two.



MARINER IV PICTURE NO. 6

Viewed with data block at left, North is at top. Sun is 22° from the zenith, from the North in the photo.

Time Taken: 5:24:09 p.m., PDT, July 14, 1965.

Slant Range: 8700 miles.

Area Covered: East-West: 190 miles. North South: 200 miles.

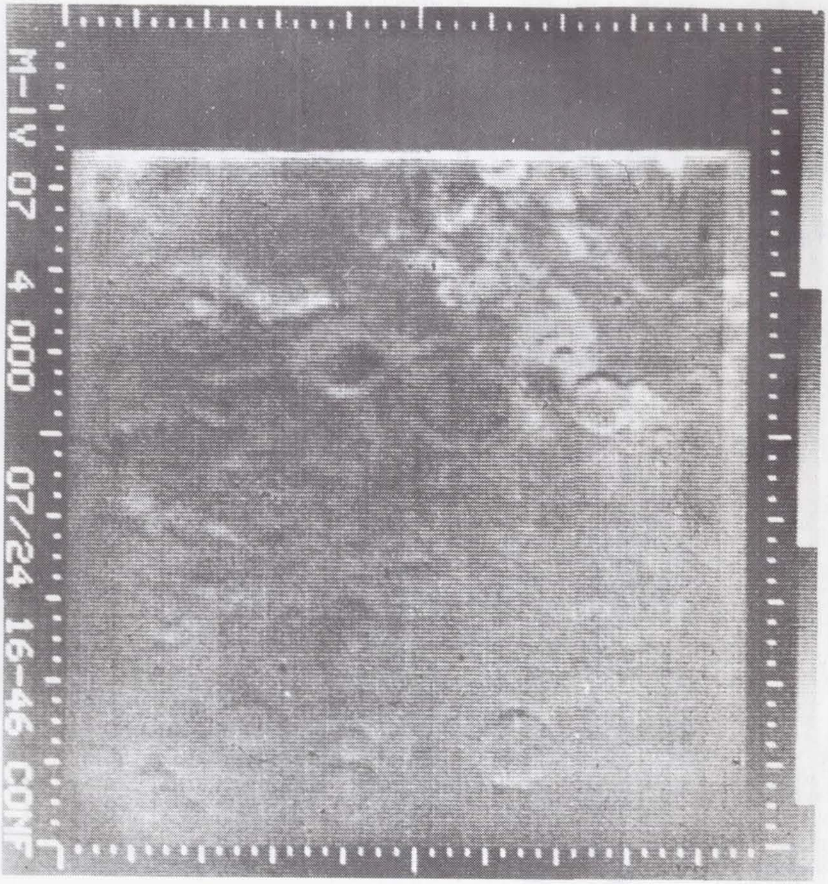
Location: 6° South Latitude, 183° East Longitude.

Map Description: Bright region in eastern Zephyria.

Filter: Green

Overlap: Upper left corner overlaps picture number 5.

Contrast Enhancement Factor: Two.



MARINER IV PICTURE NO. 7

Viewed with data block at left, North is at top. Sun is 29° from the zenith, from the North in the photo.

Time Taken: 5:25:45 p.m., PDT, July 14, 1965.

Slant Range: 8400 miles.

Area Covered: East-West: 180 miles. North-South: 180 miles.

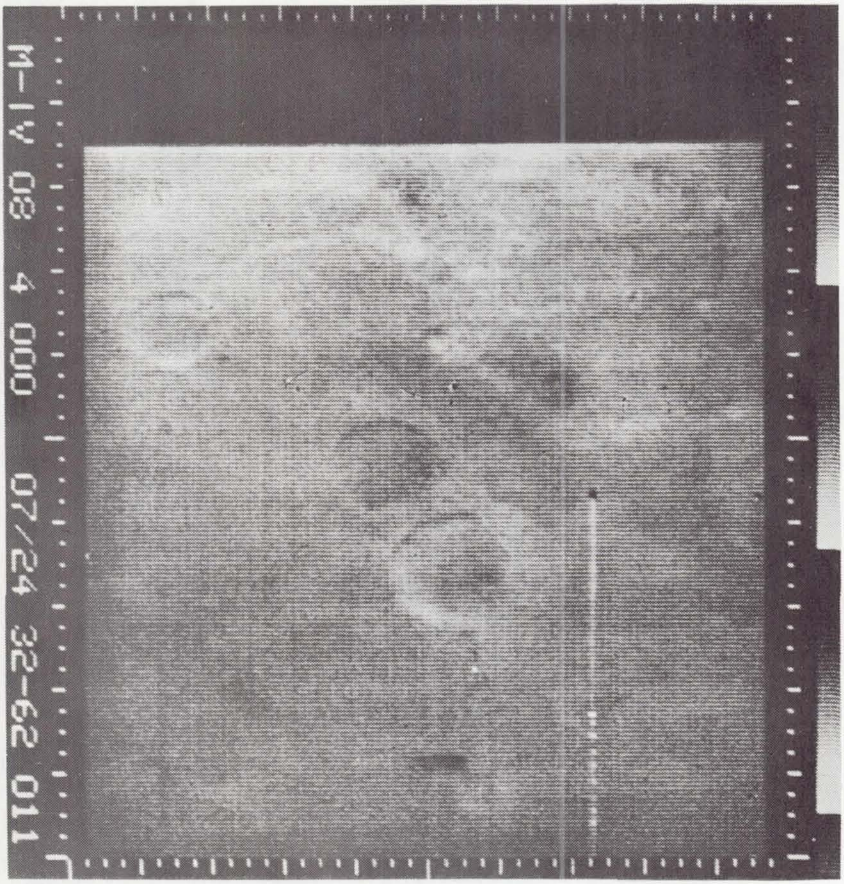
Location: 13° South Latitude, 186° East Longitude.

Map Description: Bright region in southeastern Zephyria, near Mare Sirenum.

Filter: Green.

Overlap: Lower right corner overlaps pictuer number 8.

Contrast Enhancement Factor: Two.



MARINER IV PICTURE NO. 8

Viewed with data block at left, North is at top. Sun is 32° from the zenith, from the North in the photo.

Time Taken: 5:26:33 p.m., PDT, July 14, 1965.

Slant Range: 8300 miles.

Area Covered: East-West: 180 miles. North-South: 170 miles.

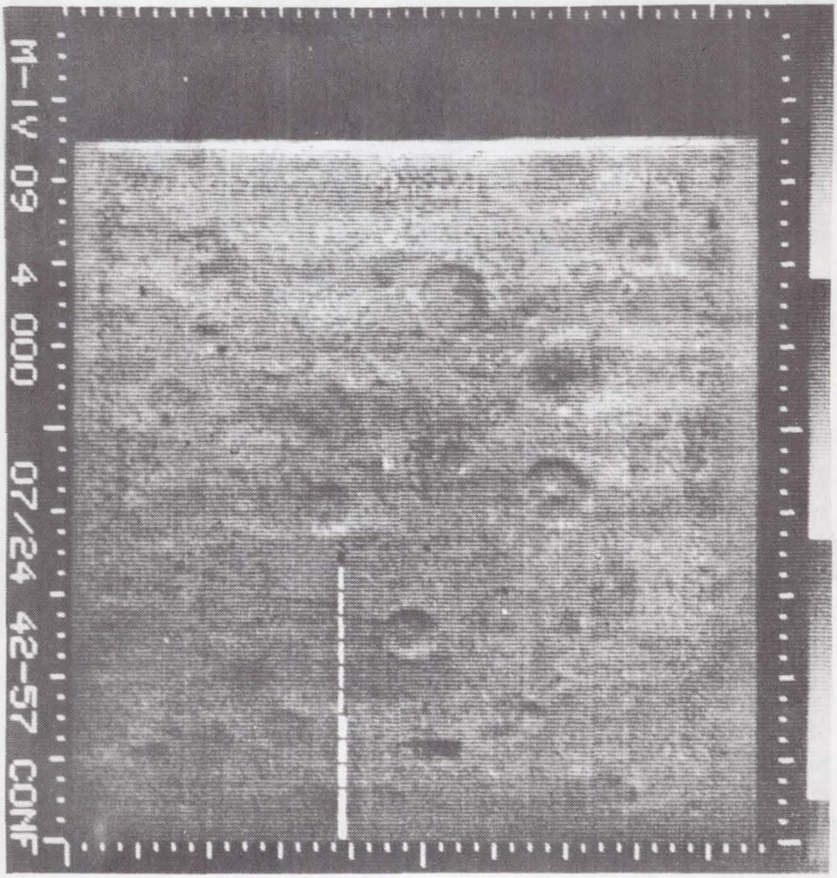
Location: 16° South Latitude, 187° East Longitude.

Map Description: Border between Zephyria and Mare Sirenum.

Filter: Orange.

Overlap: Upper left corner overlaps picture number 7.

Contrast Enhancement Factor: Two.



MARINER IV PICTURE NO. 9

Viewed with data block at left, North is at top. Sun is 38° from the zenith, from the North in the photo.

Time Taken: 5:28:09 p.m., PDT, July 14, 1965.

Slant Range: 8100 miles

Area Covered: East-West: 170 miles. North-South: 160 miles.

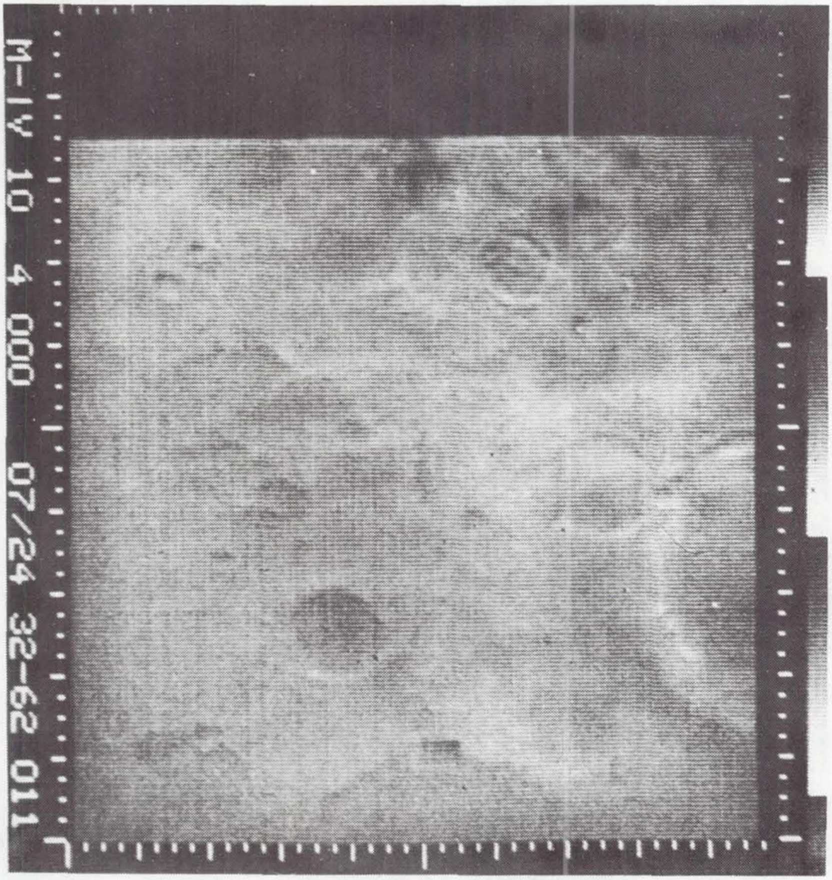
Location: 23° South Latitude, 191° East Longitude.

Map Description: Mare Sirenum, bordering on Atlantis in the southwest corner of the frame.

Filter: Orange.

Overlap: Lower right corner overlaps pictuer number 10.

Contrast Enhancement Factor: Four.



MARINER IV PICTURE NO. 10

Viewed with data block at left, North is at top. Sun is 41° from the zenith, from the North in the photo.

Time Taken: 5:28:57 p.m., PDT, July 14, 1965.

Slant Range: 8000 miles.

Area Covered: East-West: 170 miles. North-South: 160 miles.

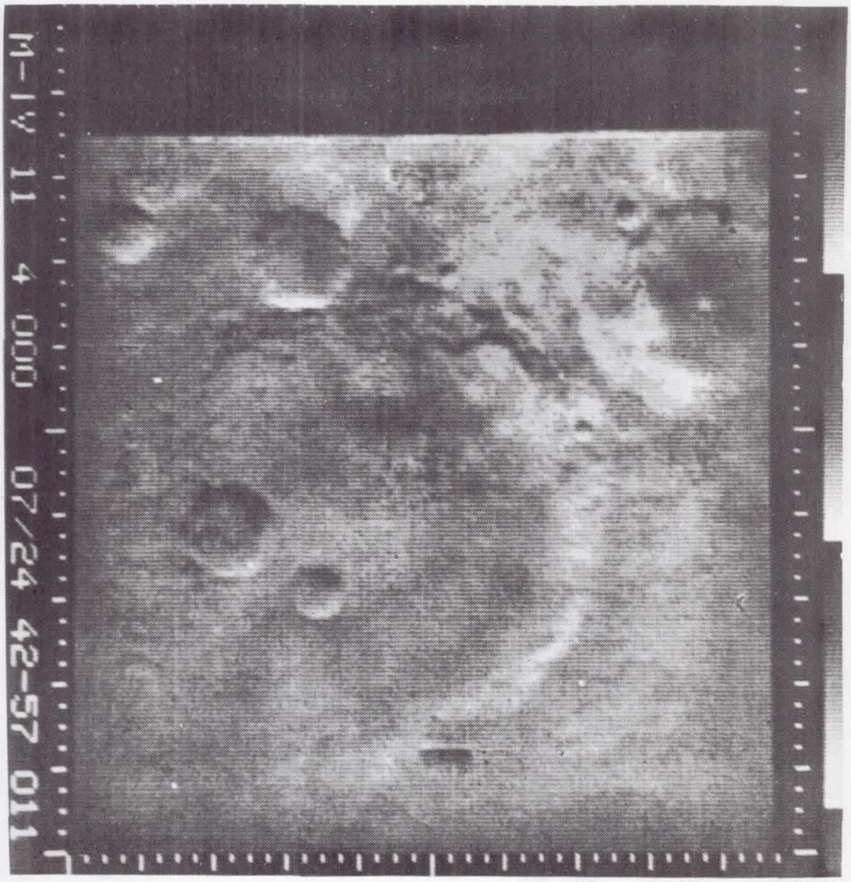
Location: 26° South Latitude, 192° East Longitude.

Map Description: Atlantis, bordering on Mare Sirenum in the northeast corner of frame.

Filter: Green.

Overlap: Upper left corner overlaps picture number 9.

Contrast Enhancement Factor: Two.



MARINER IV PICTURE NO. 11

Viewed with data block at left, North is at top. Sun is 47° from the zenith, from the North in the photo.

Time Taken: 5:30:33 p.m., PDT, July 14, 1965.

Slant Range: 7800 miles.

Area Covered: East-West: 170 miles. North-South: 150 miles

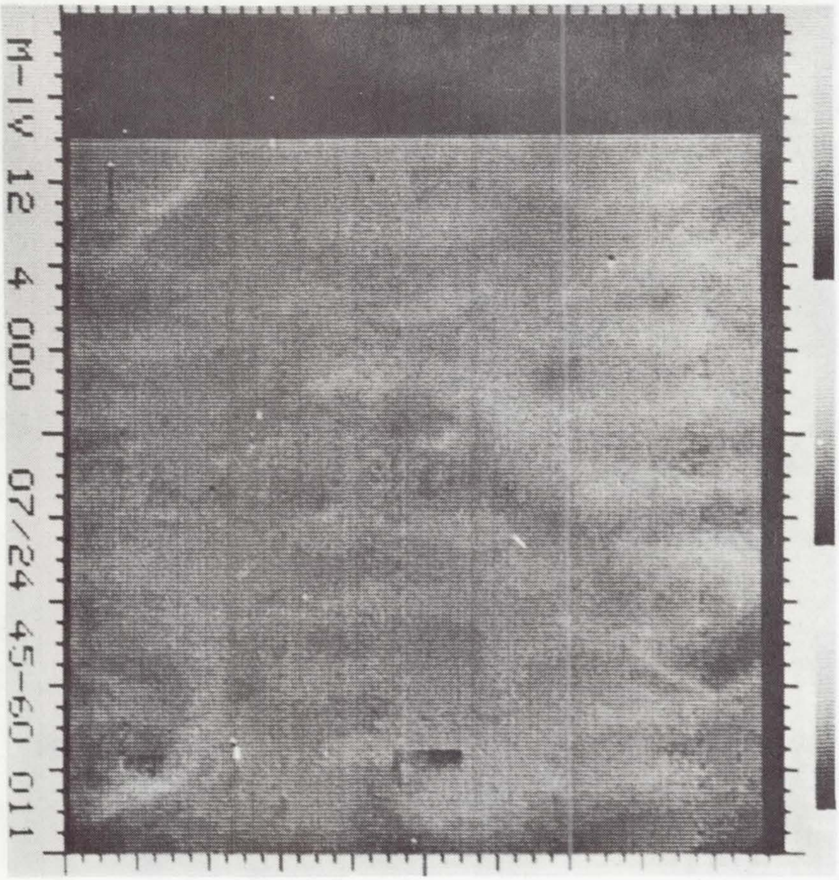
Location: 31° South Latitude, 197° East Longitude.

Map Description: Atlantis, between Mare Sirenum and Mare Cimmerium.

Filter: Green.

Overlap: Lower right corner overlaps pictuer number 12.

Contrast Enhancement Factor: Four.



MARINER IV PICTURE NO. 12

Viewed with data block at left, North is at top. Sun is 50° from the zenith, from the North in the photo.

Time Taken: 5:31:21 p.m., PDT, July 14, 1965.

Slant Range: 7700 miles.

Area Covered: East-West: 170 miles. North-South: 150 miles.

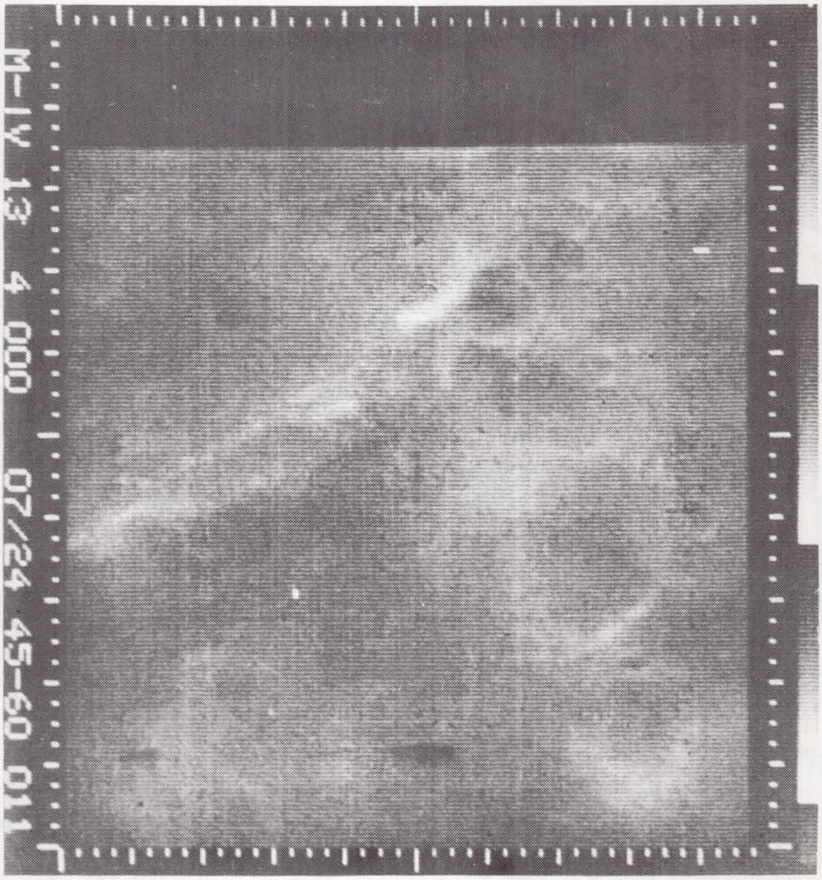
Location: 34° South Latitude, 199° East Longitude.

Map Description: Mare Cimmerium, bordering on Atlantis in the northeast corner of the frame.

Filter: Orange.

Overlap: Upper left corner overlaps picture number 11.

Contrast Enhancement Factor: Four.



MARINER IV PICTURE 13

Viewed with data block at left, North is at top. Sun is 57° from the zenith, from the North in the photo.

Time Taken: 5:32:57 p.m., PDT, July 14, 1965.

Slant Range: 7600 miles.

Area Covered: East-West: 170 miles. North-South: 140 miles.

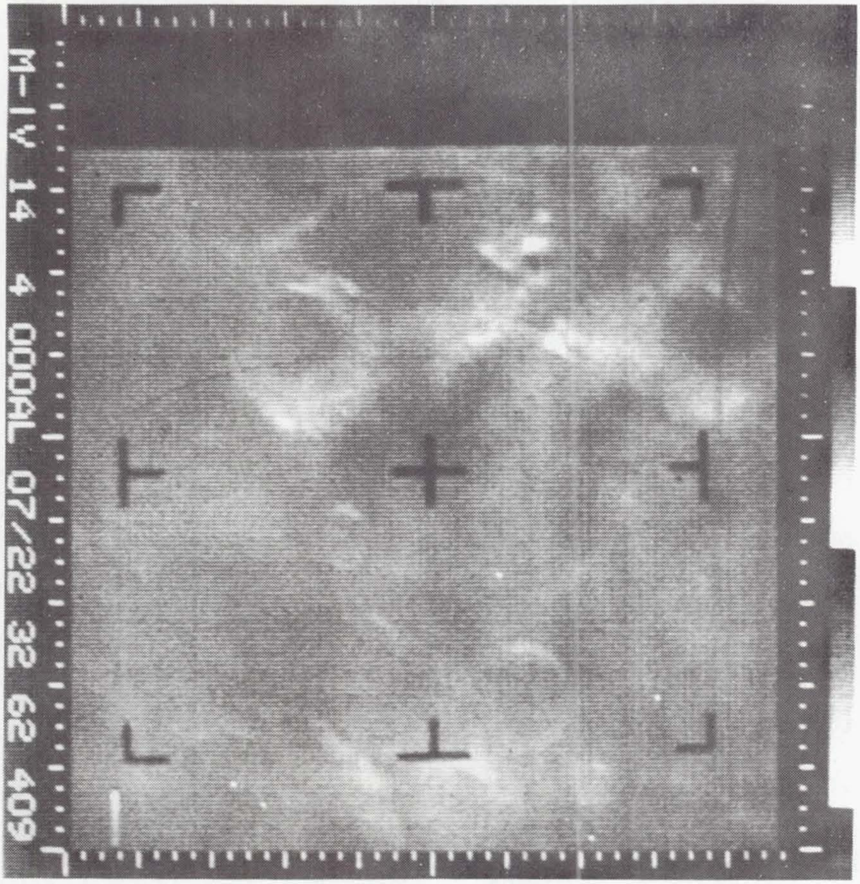
Location: 39° South Latitude, 205° East Longitude.

Map Description: Border between Mare Cimmerium to the North and the bright region Phaethontis.

Filter: Orange.

Overlap: Lower right corner overlaps picture number 14.

Contrast Enhancement Factor: Four.



MARINER IV PICTURE NO. 14

Viewed with data block at left, North is at top. Sun is 60° from the zenith, from the North in the photo.

Time Taken: 5:33:45 p.m. PDT, July 14, 1965.

Slant Range: 7600 miles.

Area Covered: East-West: 170 miles. North-South: 140 miles.

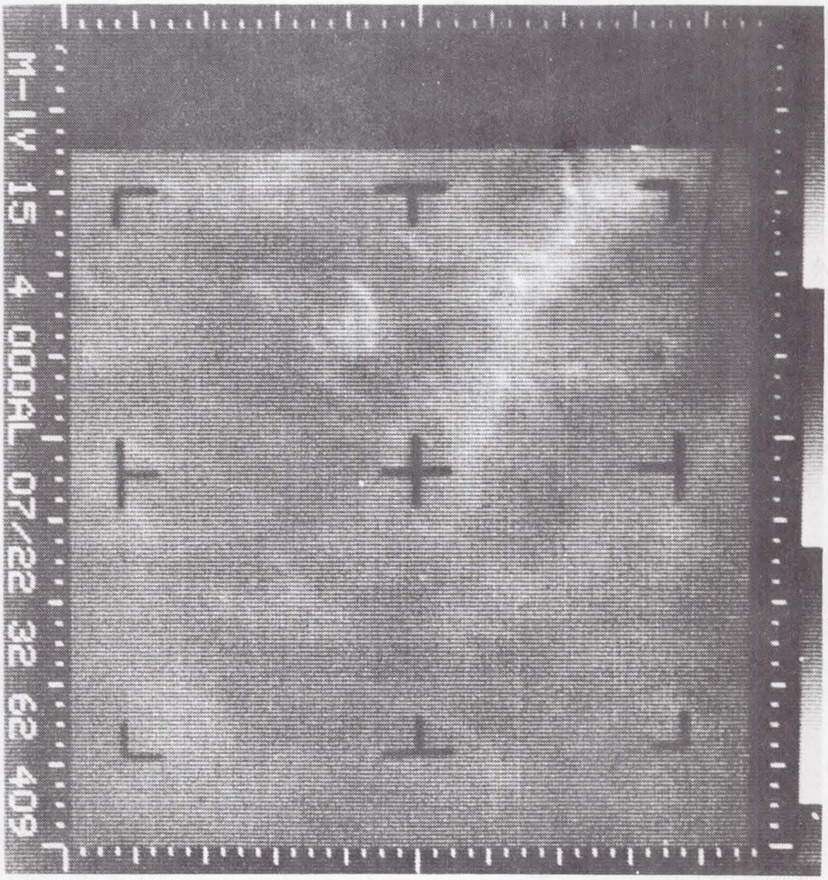
Location: 41° South Latitude, 208° East Longitude.

Map Description: Bright region, northwestern Phaethontis.

Filter: Green.

Overlap: Upper left corner overlaps picture number 13.

Contrast Enhancement Factor: Two.



MARINER IV PICTURE NO. 15

Viewed with data block at left, North is at top. Sun is 66° from the zenith, from the North in the photo.

Time Taken: 5:35:21 p.m., PDT, July 14, 1965.

Slant Range: 7500 miles.

Area Covered: East-West: 180 miles. North-South: 140 miles.

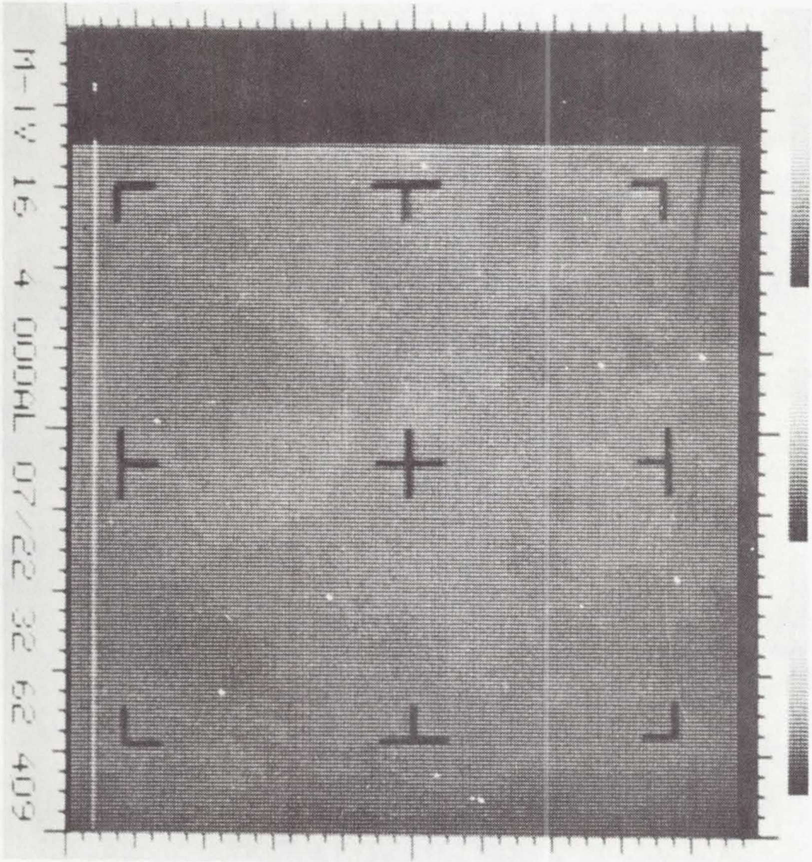
Location: 45° South Latitude, 216° East Longitude.

Map Description: Bright region in Phaethontis.

Filter: Green.

Overlap: Lower right corner overlaps picture number 16.

Contrast Enhancement Factor: Two.



MARINER IV PICTURE NO. 16

Viewed with data block at left, North is at top. Sun is 69° from the zenith, from the North in the photo.

Time Taken: 5:36:09 p.m., PDT, July 14, 1965.

Slant Range: 7500 miles.

Area Covered: East-West: 190 miles. North-South: 140 miles.

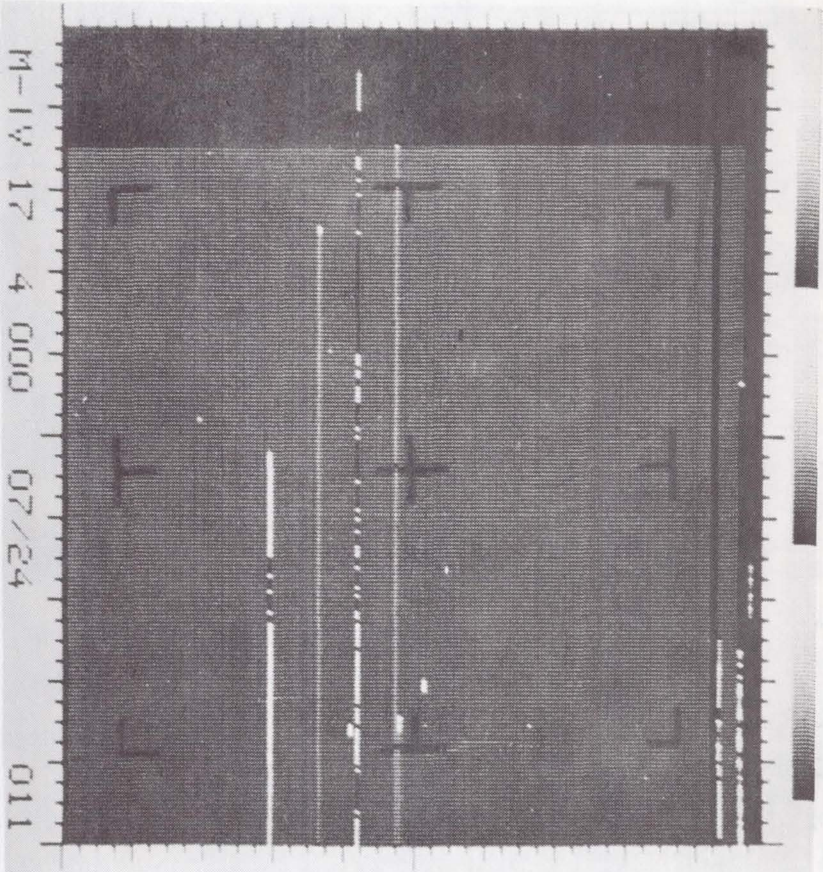
Location: 47° South Latitude, 221° East Longitude.

Map Description: Bright region in Phaethontis, near Aonius Sinus.

Filter: Orange.

Overlap: Upper left corner overlaps picture number 15.

Contrast Enhancement Factor: Two.



MARINER IV PICTURE NO. 17

Viewed with data block at left, North is at the upper right. Sun is 76° from the zenith, from the northwest in the photo.

Time Taken: 5:37:45 p.m., PDT, July 14, 1965.

Slant Range: 7400 miles.

Area Covered: Northeast-Southwest: 200 miles. Northwest-Southeast: 140 miles.

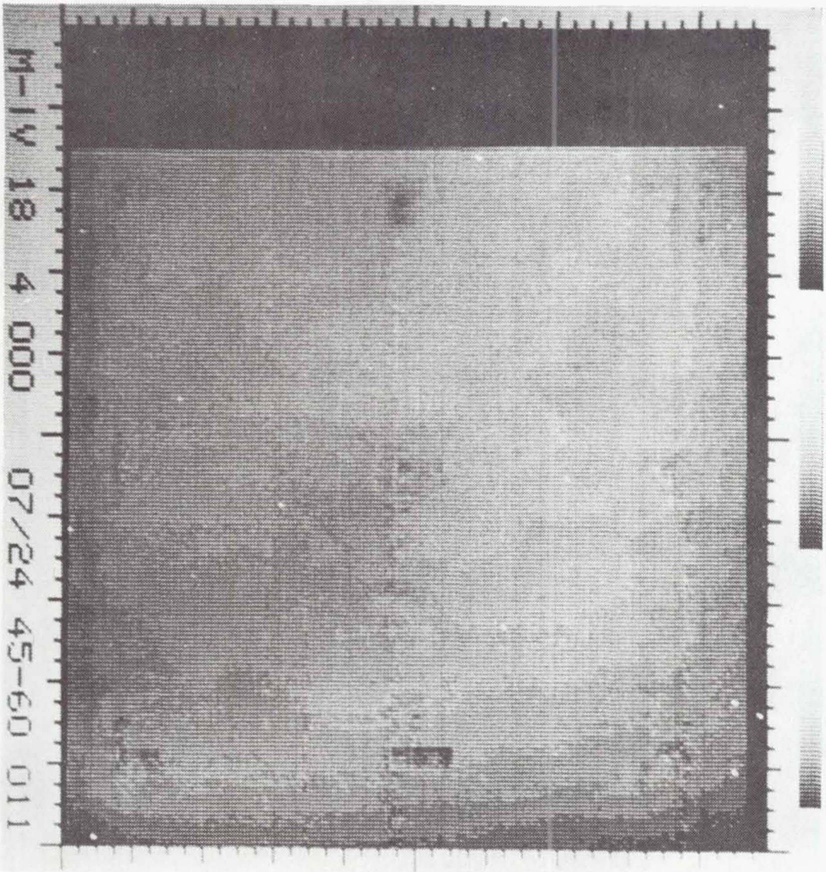
Location: 50° South Latitude, 232° East Longitude.

Map Description: Dark region in Aonius Sinus.

Filter: Orange.

Overlap: Lower right corner overlaps picture number 18.

Contrast Enhancement Factor: This picture is in raw form with no enhancement.



MARINER IV PICTURE NO. 18

Viewed with data block at left, North is at the upper right. Sun is 80° from the zenith, from the northwest in the photo.

Time Taken: 5:38:33 p.m., PDT, July 14, 1965.

Slant Range: 7400 miles.

Area Covered: Northeast-Southwest: 210 miles. Northwest-Southeast: 140 miles.

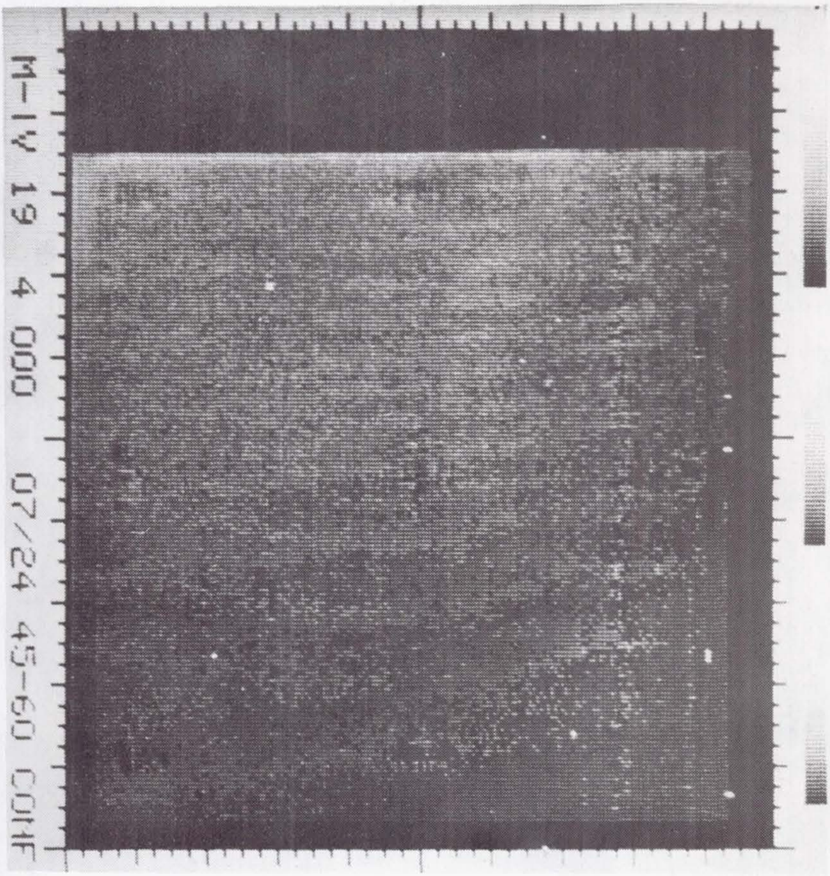
Location: 51° South Latitude, 238° East Longitude.

Map Description: Dark region in Aonius Sinus.

Filter: Green.

Overlap: Upper left corner overlaps picture number 17.

Contrast Enhancement Factor: Four.



MARINER IV PICTURE NO. 19

Viewed with data block at left, North is at the upper right. Sun is 88° from the zenith, from the northwest in the photo.

Time Taken: 5:40:09 p.m., PDT, July 14, 1965.

Slant Range: 7500 miles.

Area Covered: Northeast-Southwest: 240 miles. Northwest-Southeast: 150 miles.

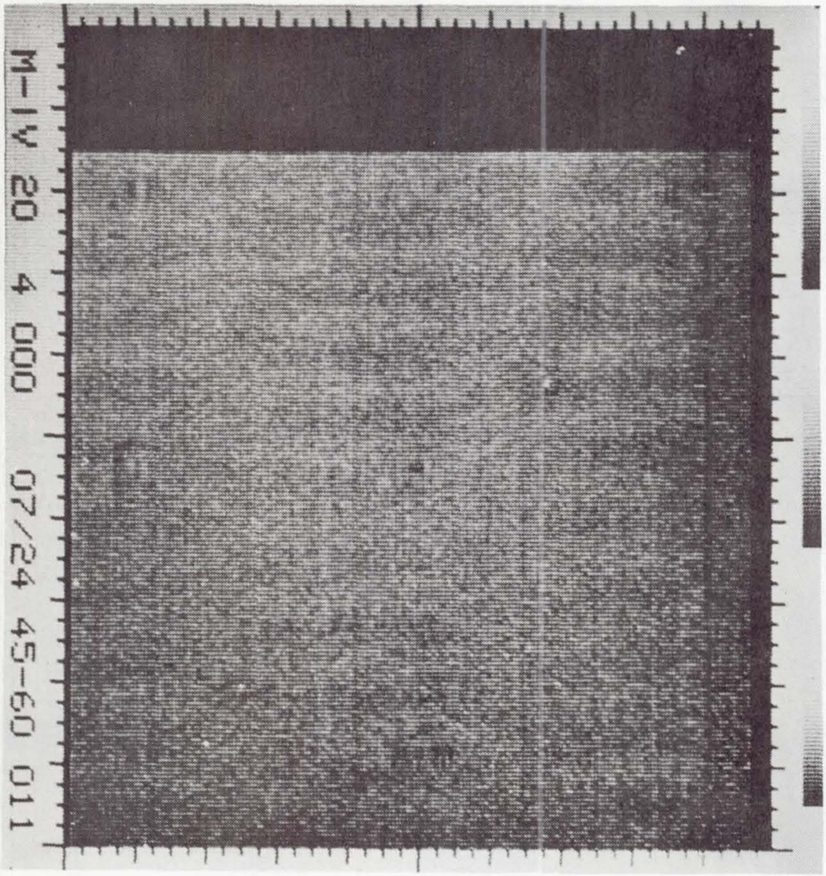
Location: 51° South Latitude, 253° East Longitude.

Map Description: DArk region in Aonius Sinus. Terminator in eastern corner of frame.

Filter: Geeen.

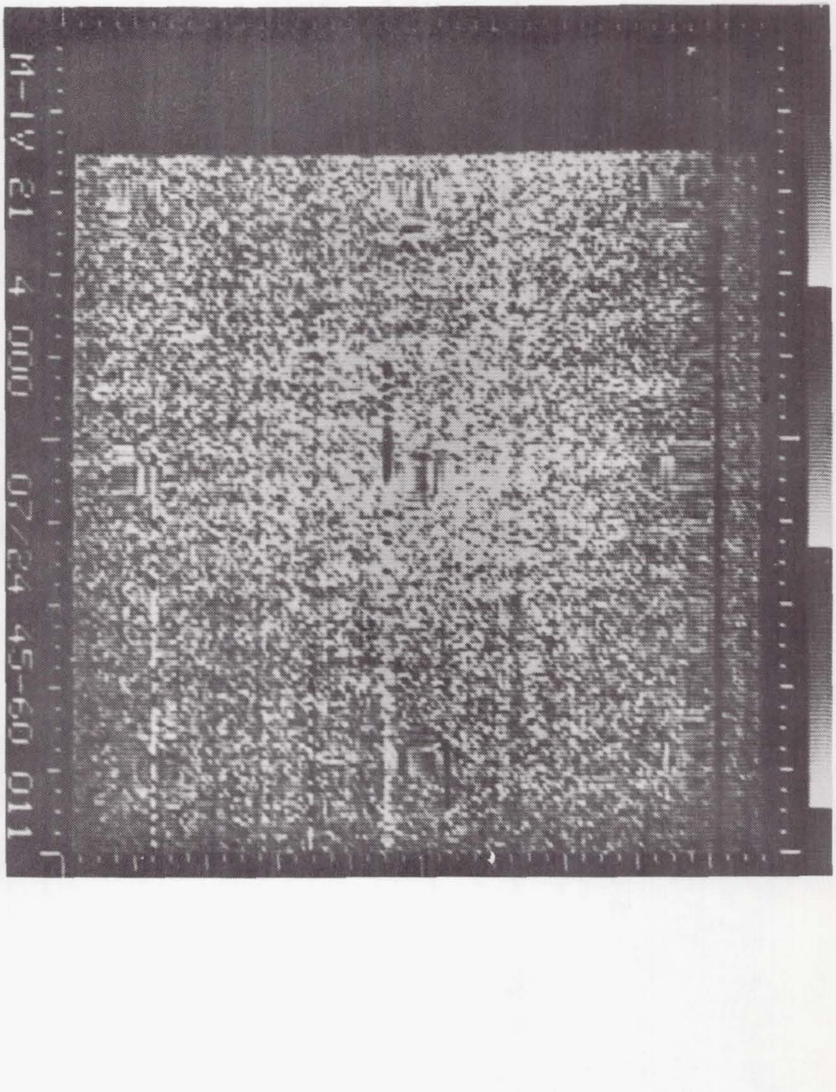
Overlap: Lower right corner overlaps picture number 20 in the terminator region.

Contrast Enhancement Factor: Four.



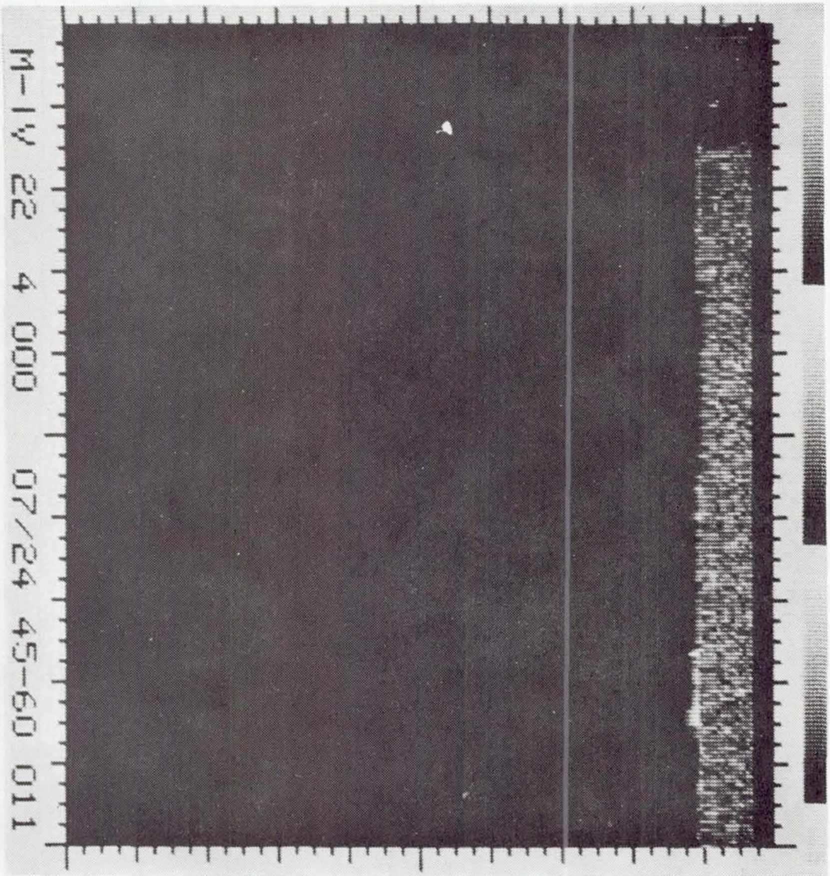
MARINER IV PICTURE NO. 20

This frame is almost entirely beyond the terminator. Its upper lift corner overlaps picture number 19 in the terminator region.



MARINER IV PICTURE NO. 21

This frame is entirely beyond the terminator.



MARINER IV PICTURE NO. 22

This partial frame is entirely beyond the terminator, and may be partly beyond the dark limb of the planet.

N66 31496

GOALS OF THE VOYAGER PROGRAM

By

Robert Fellows

NASA - Office of Space Science

and Applications

INTRODUCTION

Voyager is a program for the unmanned scientific exploration of the planets. It will continue and extend the results of the Mariner Program and represents the intention of the National Aeronautics and Space Administration to accelerate the pace of planetary exploration.

The roots of Voyager can be traced back to early 1961 when studies were begun of larger spacecraft with capabilities significantly greater than those of the Mariner Program, which was then underway. Mariner II later performed a successful flyby mission of Venus in 1962, and Mariner IV has just recently completed a spectacular flyby and photo-reconnaissance of Mars.

One objective of these early studies was to determine the capabilities a spacecraft should have to perform detailed exploration and characterization of another planet. These studies were continued during the design and development of Mariner II and Mariner IV, and were originally directed at the 1966 and 1969 launch opportunities to Mars.

It appeared from these studies that a spacecraft in the 6,000 to 8,000 pound range was a logical step, although later studies continued to consider both higher and lower weight classes to appraise capability versus cost, and to determine the bounds of technical feasibility.

THE VOYAGER PROGRAM

The various types of scientific investigations to be conducted when exploring a new planet are summarized in Figure 1, which lists the objectives a spacecraft system and its scientific payload should be capable of achieving. A not unexpected result of these early studies was a requirement for high payload capability if significant, detailed measurements of the environment, atmosphere, and surface of the planet are to be obtained. A second requirement is that the system have a long lifetime at Mars; that is, the spacecraft should operate on the surface of the planet or in orbit about the planet for fairly long periods of time. This is particularly important for Mars since seasonal variations have been noted and observations of these variations may be particularly informative. The third requirement follows directly from the first two, i. e., capability to handle a large amount of data. In addition to handling the output of the scientific instruments, high-quality visual data are necessary, both from orbit and from the Martian surface.

Figure 2, summarizes the results of these early Voyager studies. Four different types of missions are shown. The early mission is the Flyby, such as Mariner II and IV. The next step in remote observations is the Orbiter. (Orbiter spacecraft weight on this chart includes the retropropulsion equipment and fuel necessary to achieve orbit about the planet.) Two classes of landed missions are indicated. "Probes/ Capsules" are those with battery power systems, while "Lander/Capsules" are larger and would probably contain nuclear power systems.

Large increases in observational lifetimes are available with orbiter and lander missions. Once certain spacecraft weight thresholds have been crossed, the effect of lifetime on spacecraft weight is relatively small. The studies also show that 10^8 to 10^{10} bits of data are available for orbiter and lander missions. Again, the effect of spacecraft weight is reasonably small. Thus, the size of the system is essentially established by the weight desired for scientific experiments.

For orbiter missions, about 200 to 300 pounds of science payload appears consistent with the scientific observations to be done. Consequently, about 3,000 to 5,000 pounds (including retropropulsion) will suffice for the orbiter portion of the mission.

The weight of scientific instruments carried by landers turns out to be about 5 to 10% of the weight of the lander or capsule as it enters the Martian atmosphere. Recognizing that numerous experiments in different scientific disciplines are to be performed, the best estimates today indicate a minimum weight of about 100 pounds. For a long-term program, this weight should increase. It has even been suggested by some that an ultimate landed instrument weight of 2,000 to 5,000 pounds is desirable. This would require extremely heavy landers, about 20,000 pounds or more. Landers up to about 6,000 pounds appear feasible for the next ten years or so.

The studies providing these results have led to the current Voyager concept shown in Figure 3. The spacecraft consists of three modules: A basic bus, which is the octagonal ring to which the solar panels are attached; a retropropulsion system to place the bus into orbit; and a large landing capsule. With these three modules, Voyager can do both orbiter and lander missions. Voyager is currently planned to have a total weight of from 7,000 to 10,000 pounds, sized for the Saturn IB/Centaur launch vehicle. This size will provide the desired scientific capability for a detailed Martian exploration program.

The basic Voyager orbiter, as shown in Figure 4, will weigh about 2,000 pounds and will be suitable for either flyby or orbiting missions. It will be eight or nine feet in diameter and will carry 200 to 300 pounds of scientific instruments. The technologies of the bus subsystems are essentially Mariner state of the art, although improvements will be sought, particularly in the data storage and communications subsystems. The data automation equipment will also be more complex because of the larger number of science instruments to be accommodated.

The advanced mission studies conducted in the past few years have shown that a spacecraft bus of the Voyager class designed for Mars missions will also have direct application to Venus missions. Some modifications will be required, particularly in the area of thermal control. Many of the spacecraft's technologies and subsystems will also have direct application to more difficult and advanced missions, such as Jupiter flybys.

A typical retropropulsion module, to be used for placing the bus in orbit about Mars, may use space-storable propellants at a thrust level of about 1,500 to 2,000 pounds. Variable amounts of propellant will be necessary for different missions depending upon the launch opportunity and the type of orbit desired.

The entry capsule may have a shape similar to the Apollo Command Module as shown in Figure 5. However, other shapes are still under study. Capsule weight may range from about 1,000 pounds early in the program to 6,000 pounds for later missions. One of the unique technological characteristics of the capsule is that radioisotope thermal generator (TRG) power supply systems may be needed for long lifetime on the Martian surface. The sterilization requirements for a Mars lander must also be satisfied.

A possible program evolution for Mars missions is shown in Figure 6. It could be as follows: test flights of the bus in 1969; the first operational mission in 1971 involving a 1,800 pound bus placed into orbit by a 3,000 pound retropropulsion unit. The orbit assumed here is 1,000 kilometer perigee and 4,000 kilometer apogee. This raises the total weight to around 5,000 pounds, permitting several thousand pounds for the lander. However, it is anticipated that the first landing capsule will be less, possibly 1,000 to 2,000 pounds. If this approach is followed, the 1973 missions could be an orbiter and a larger capsule. By 1975, a flyby with a capsule as large as 5,000 to 6,000 pounds would be reasonable. Throughout the program, the same bus would be used, both for orbiting and flyby missions. It should be pointed out that while orbits as close as 1,000 kilometers are assumed, higher orbits, possibly to 4,000 kilometers perigee, may be required depending on the density of the upper atmosphere of Mars. The choice of orbit is influenced by a requirement for long orbital life if the spacecraft is not sterilized. Present plans are that the orbiter will not be sterilized.

Sterilization of the capsule is a definite requirement in keeping with the United States policy of not contaminating Mars by organisms from Earth. Current requirements indicate that all instruments, materials, and the flight capsule itself must be capable of surviving a terminal heat sterilization treatment at 135°C for 24 hours. Cleanroom assembly will also be required. Qualification testing of scientific instruments and capsule subsystems will probably be three, 36-hour heat cycles at 145°C. Thus, all parts of the capsule including scientific instruments will be designed to withstand rigorous thermal cycles while still retaining a high reliability and long operating life. Rigorous manufacturing, assembly, and handling

procedures will be required; undoubtedly resulting in complex manufacturing, testing and check out methods.

In addition to sterilization, other major problem areas will be defined in detail during design definition studies which will include communications and data storage, nuclear power supplies, and retropropulsion.

The large increase in total data, and the higher rate at which it will be acquired, will require much larger data storage capability than the Mariner spacecraft possessed. Higher wattage power amplifier tubes and large directional antennas on the orbiter will be required.

Nuclear power supplies may be the solution to long-life power supply problems for the landed capsule; but, with the attendant introduction of problems in radiation shielding, spacecraft and capsule thermal control, and pre-launch assembly and handling, the retropropulsion system necessary for placing the spacecraft in orbit about the planet must be capable of operating reliably after storage in space for seven or more months.

At this stage, it should be interesting to review quickly the capabilities of Mariner IV and compare them with the goals set for the Voyager system. Figure 7, indicates the main features of Mariner IV. Of particular interest for comparison with the Voyager spacecraft are the gross weight of 575 pounds versus a 2,000 pound orbiter plus a 2,000 to 6,000 pound capsule; eight experiments totalling 60 pounds versus a 200 pound orbiting payload of possibly 20 experiments plus an initial planned capability of 50 pounds of instrumentation in the landing capsule; and a six month orbiter lifetime at Mars contrasted to a flyby mission.

Figure 8, is an artist's concept illustrating some features of the Voyages system--to place a significant payload of scientific instrumentation in orbit about a planet, and to place on the surface a payload of sufficient size and lifetime to conduct significant "in situ" exploratory scientific investigations.

Figure 9, shows an important step in this sequence. Here the artist portrays the entry of the capsule after separating from the spacecraft, and the start of repositioning of the spacecraft prior to firing of its retrorockets.

Figure 10, shows the artist's concept of the landed capsule with instruments and sample acquisition hardware deployed for operation while the far overhead orbiter is firing its retropropulsion to obtain the necessary velocity change to achieve orbit about the planet. Actually, this event will take place at a very considerable distance from the capsule landing point.

The orbiter will carry a scan platform, possibly several, to accommodate the television, spectroscopic, and other instruments requiring pointing and scanning across the planetary disc. It will also carry booms to remove magnetometers as far as possible from fields created by the spacecraft instruments and systems. Large directional antennas will be needed which may be larger than the spacecraft body itself.

The capsule will have sample acquisition and instrumentation facilities to permit the examination of the planetary surface, soil, and atmosphere. The details and capabilities of this system will be determined by the payload weight available and the specific experiments selected for each mission.

Mars has been selected as the target for the first operational mission of Voyager because, in many respects, it appears to be the planet most similar to Earth and because of the seasonal variations in surface features that have been observed and other evidence interpretable or suggestive of the possibility of indigenous life.

In October 1964, the Space Science Board of the National Academy of Sciences published a report designating "the exploration of the nearer planets as the most rewarding goal on which to focus national attention for the 10 to 15 years following manned lunar landing." The report also recommended that Mars be given first priority within the planetary program.

VOYAGER EXPERIMENTS

In advance of the actual selection of experiments for the 1971 Voyager mission, we might examine the general areas of experimentation involved in the exploration of Mars.

Figure 11, lists some of the possible experiments that the orbiter might carry. Television and facsimile scanners should be able to perform mapping in the visible and infrared spectral regions, and to detect interesting topographical features and circulation patterns of suspected dust storms. Ultraviolet and infrared spectrometers can study the atmospheric composition, and give some information on temperature, pressure, scale height, and the diurnal and seasonal variations which occur.

Microwave spectrometry and radiometry can furnish additional information about the atmosphere and surface characteristics. The gamma ray experiment would furnish information about naturally radioactive elements such as potassium 40, thorium, and uranium. An ionosphere sounder can study the Martian ionosphere from above in the same way that satellite-borne topside sounders have studied the Earth's atmosphere.

An early orbiter performing exploratory measurements should also carry a magnetometer and energetic particle detectors to study not only the trapped radiation that may exist around Mars but also to monitor cosmic ray and solar proton energy inputs to the planet.

A micrometeorite detector would be needed both as a science experiment and also to furnish data for the planning and design of spacecraft and hardware for future missions. How much greater is the flux in the vicinity of Mars than in interplanetary space or that near the Earth? A gravimeter carried by an orbiter would furnish information concerning the distribution of mass of the planet, or variations in surface crustal density.

Notice that the weight totals 271 pounds. It is very doubtful that all of these experiments would go on the first mission.

Figure 12, shows a similar list of experiments which have been proposed and discussed for use in a capsule mission. These can be divided into two groups. One would determine and characterize the physical environment of the planet; the other group would be all types of experiments connected with the possibilities of extraterrestrial life. The characteristics of the atmosphere would be defined using pressure gauges, manometers, and anemometers to determine surface pressure, winds and temperature of the atmosphere. Composition could be obtained by a mass spectrometer and a gas chromatograph, either independently or working in a series arrangement. Also, at a very light weight, there is a possibility of including single-composition detectors that have a response for a specific component---for instance, an oxygen detector that would report the amount of oxygen at the Martian surface, or a water vapor detector that would report the amount of water vapor at or even below the surface.

There are also experiments to determine the characteristics of the surface soil itself. Some information on hardness can be obtained by a study of the deceleration profile of the spacecraft. A penetrometer could determine the characteristics of the surface and subsurface soil to a resonable depth. Physical characteristics of the surface such as the magnetic and electric properties are also of interest. A gamma-ray spectrometer aboard the capsule would also give information as mentioned already for the orbiter. Seismometers have been proposed, both active and passive types, to give information on the motions of the planet. Does it have "earthquakes?" What are their frequency and distribution?

There would be energetic particle detectors to determine the charge, mass, and energy of energetic particles such as cosmic rays and solar protons. Mars, having a much thinner atmosphere than the Earth, will probably have a much higher flux at the surface.

X-ray diffraction techniques can also be used to determine the surface composition and structure. Neutron analysis has been proposed. If a neutron generator could be included, one would have the opportunity of performing a number of different types of experiments such as neutron activation, and elastic and inelastic scattering techniques to determine the elemental components of the planetary surface.

A mass spectrometer could be augmented by pyrolysis techniques to examine volatiles from the decomposition of solid materials forming the surface. This represents, then, an experiment useful to planetology and the biosciences, depending upon the nature of the products. A strong indication of organic material could be valuable evidence concerning the existence at sometime of some sort of indigenous life.

The fluorescence spectrometer, UV spectrometer, J-band spectrophotometer, optical rotatory dispersion experiment, the fluorimeter, and the luciferase reaction are all spectrometric or photometric techniques designed to look for complex organic molecules.

Mass spectrometry and gas chromatography appear to be a useful combination of instruments for solids analysis also. A sample would be vaporized or pyrolyzed, and the products would be analyzed for mass distribution and identified by gas chromatography. The results would then be used to decide the nature of the original material. The microcalorimeter involves extremely sensitive calorimetry techniques to search for living material by detecting the heat energy released through metabolism. This requires extremely stable equipment and very, very sensitive techniques. An infrared spectrometer can identify or search for large, complicated organic

molecules. This assembly of experiments totals approximately 500 pounds and would require 10^8 bits of telemetry.

The first Voyager capsule will not have the capability of doing the scope of research that is portrayed here, but the long-range goal of the Voyager system is to achieve a capability as significant as this.

In addition to the experiments which would be performed at the climax of its mission, during the long months spent in transit Voyager will study the cosmic radiation, solar plasma, magnetic fields, and micrometeoroid flux of interplanetary space.

Thus, based on significant advances now available in boosters, tracking, communications, power equipment, and component reliability, Voyager represents an acceleration of the pace of planetary exploration. The National Aeronautics and Space Act of 1958 which created the National Aeronautics and Space Administration contains, among others, these objectives: "To expand our knowledge of phenomena in space and the atmosphere, and to preserve the role of the United States as a leader in aeronautical and space sciences and technology, and in the application of these disciplines to peaceful work here on Earth and in space." In following these objectives, the Voyager Program extends a major challenge and great opportunity to the scientific community and to the aerospace industry.

ADDENDUM -- JANUARY 1, 1966

In late December 1965, the National Aeronautics and Space Administration announced that the first Voyager mission had been deferred until 1973 and that one Mariner flight to Venus in 1967 and two Mariner flights to Mars in 1969 had been added to the program.

The flight spacecraft which backed up the highly successful Mariner IV mission to Mars in 1965 will be modified for the Venus mission. The scientific experiments will be improved versions of those flown on the previous Mariners. A somewhat heavier Mariner based on the technology of the previous Mariners will be developed for the missions to Mars in 1969. The Atlas/Centaur launch vehicle will be used.

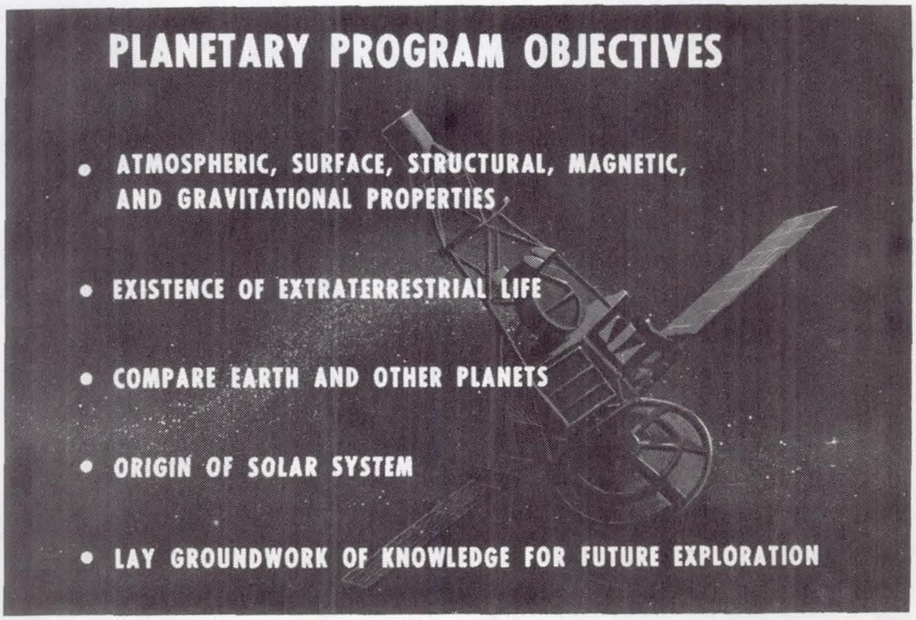


FIGURE 1

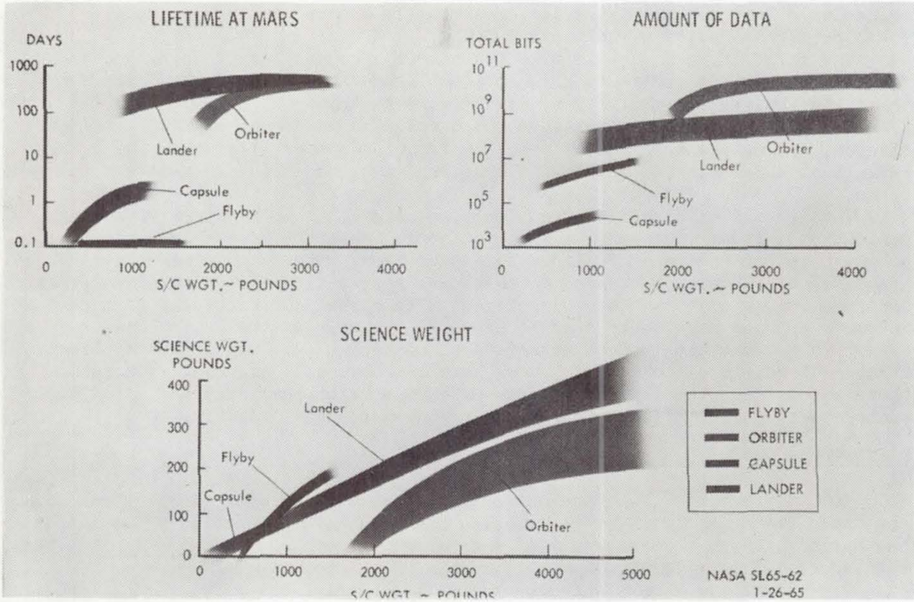


FIGURE 2

Spacecraft Capability

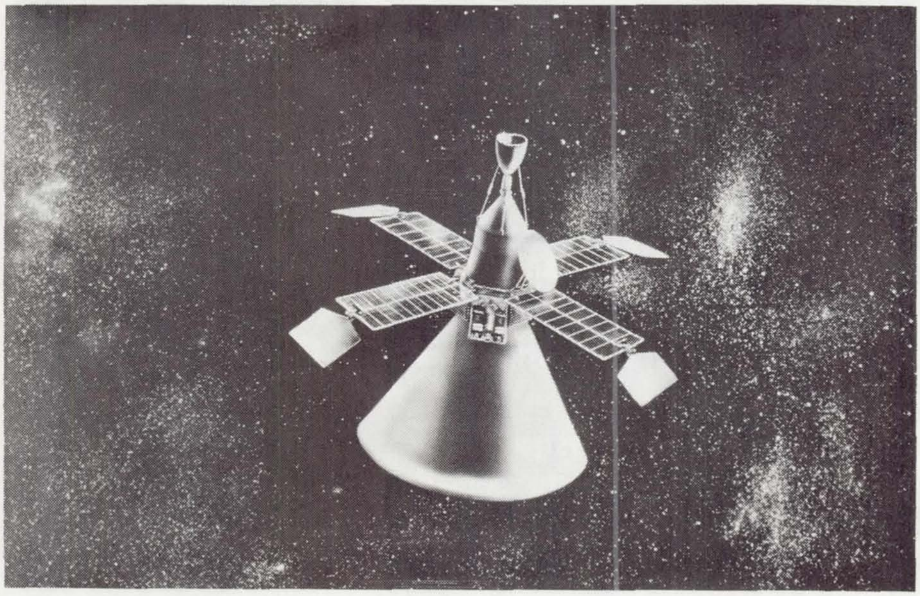


FIGURE 3

The Voyager Spacecraft

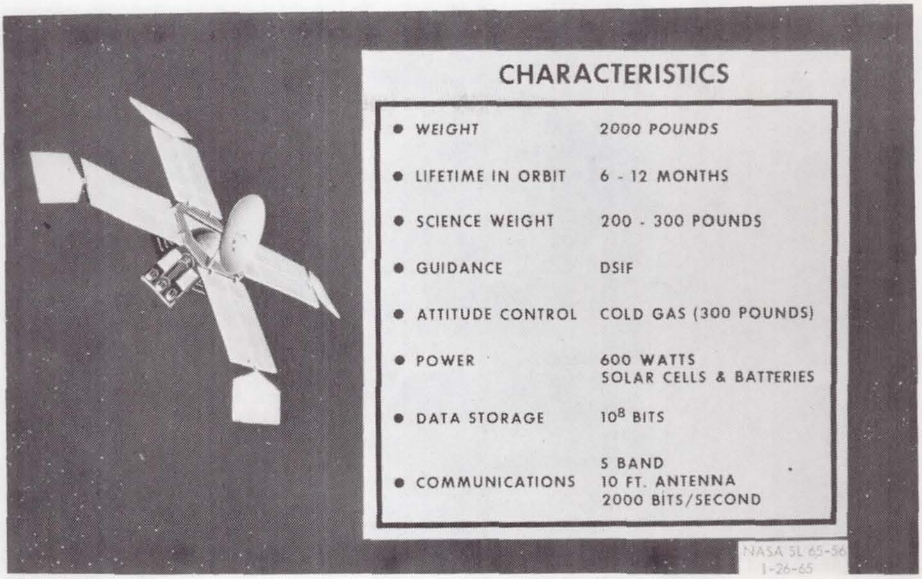


FIGURE 4

Voyager Bus

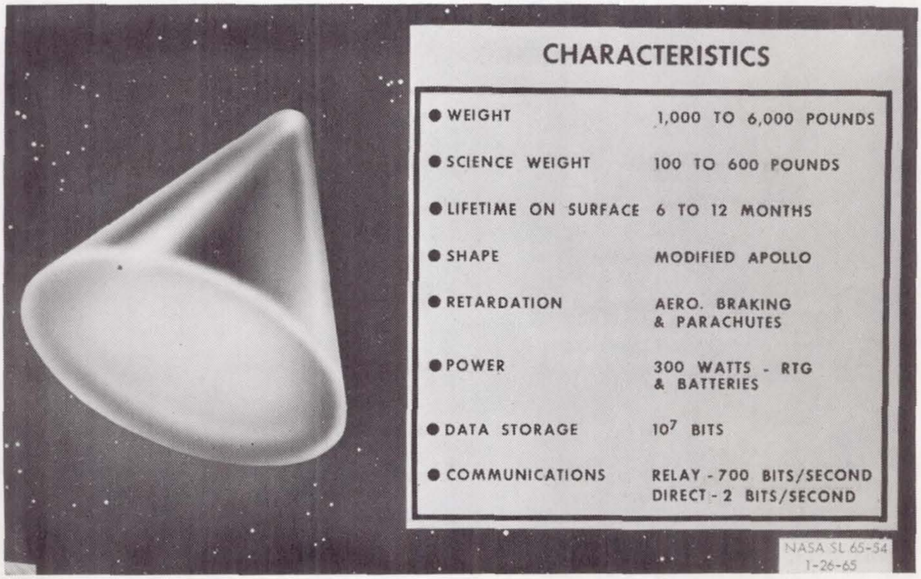


FIGURE 5

Voyager Capsule

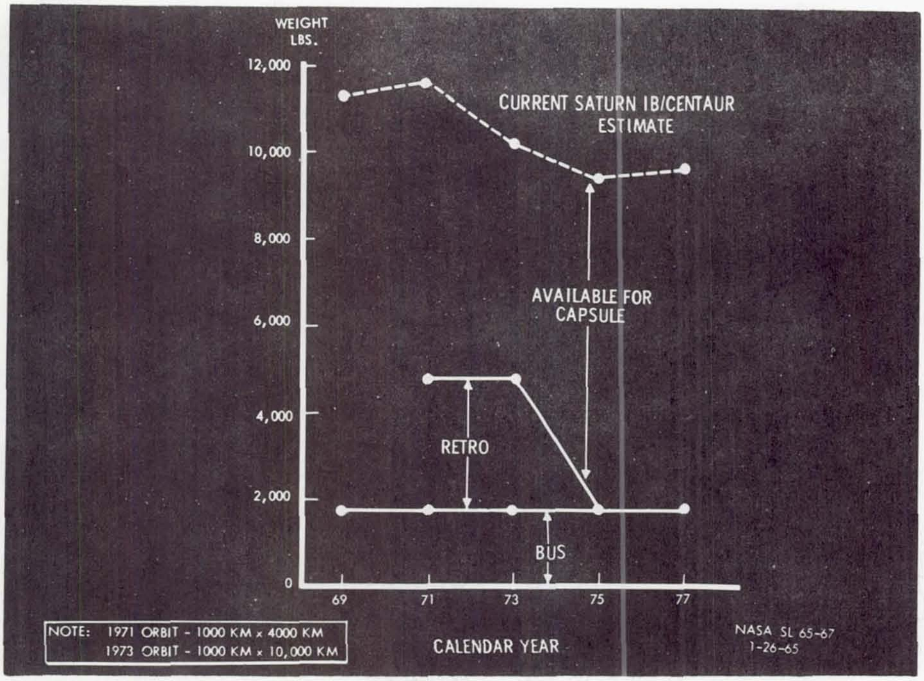


FIGURE 6

Program Evolution

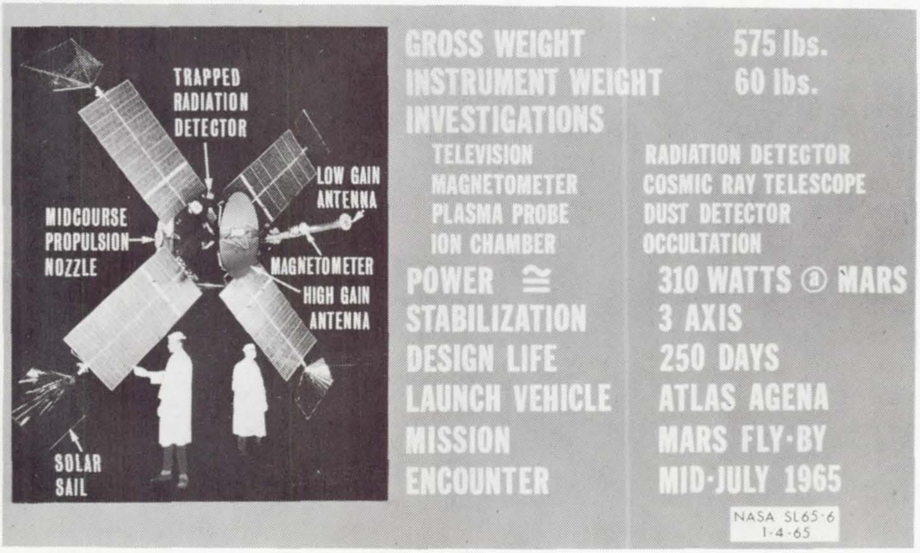


FIGURE 7

Mariner Mars - 1964

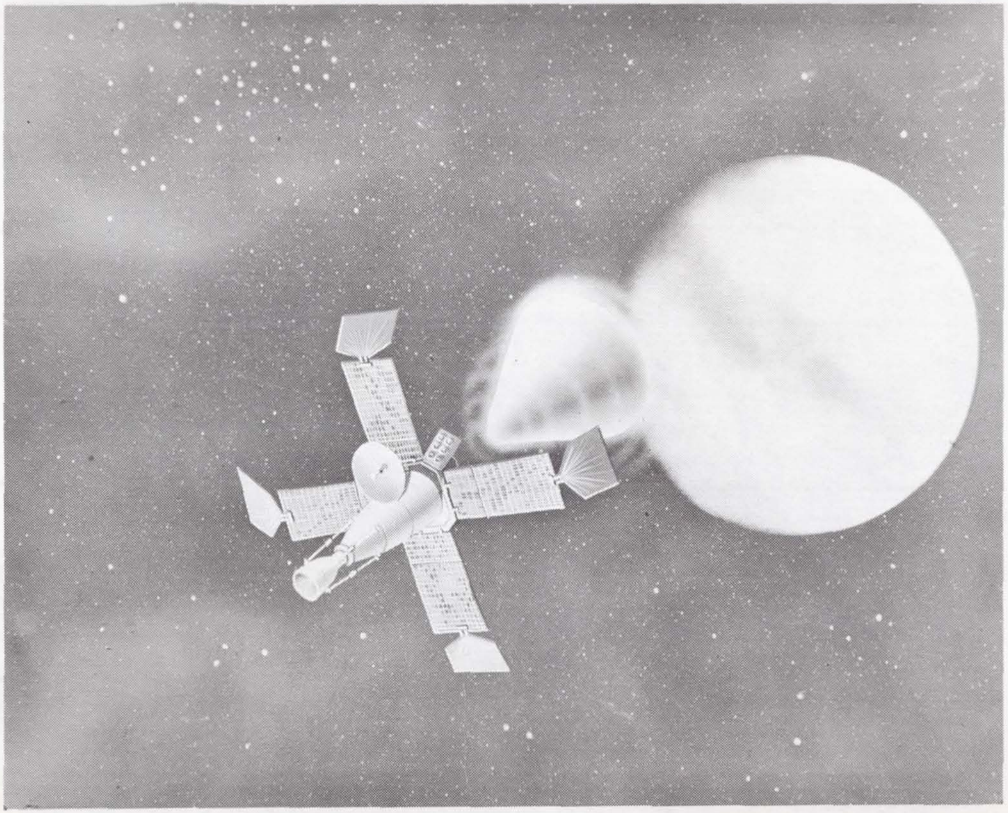


FIGURE 8

Artist's concept of the Voyager System

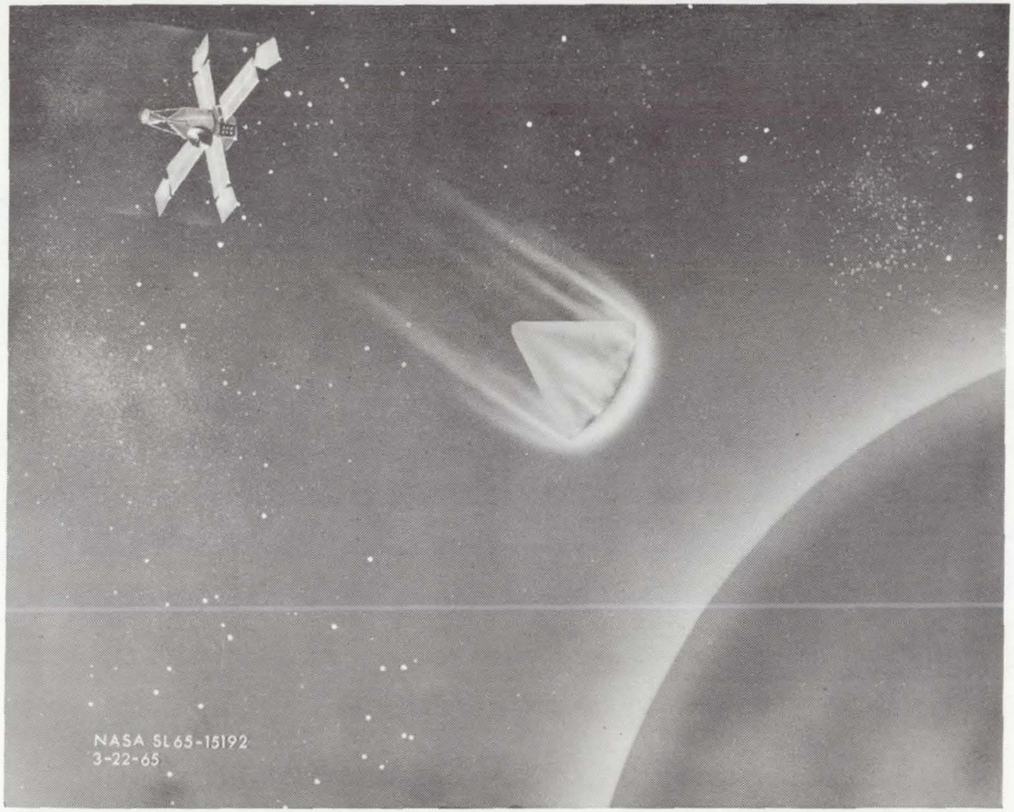


FIGURE 9

Artist's concept of the entry of the capsule after separating from the spacecraft

FIGURE 10

Artist's concept of the landed capsule with orbiter above

INSTRUMENTS	WEIGHT, POUNDS
TELEVISION	130
VISUAL & IR SCANNER	15
GAMMA RAY SPECTROMETER	8
MICROWAVE SPECTROMETER	35
ULTRAVIOLET SPECTROMETER	22
INFRARED SPECTROMETER	26
IONOSPHERE SOUNDER	15
TRAPPED RADIATION DETECTOR	4
MICROMETEORITE DETECTOR	8
MAGNETOMETER	5
GRAVIMETER	3
	271

TOTAL DATA = 10^{10} BITS
LIFE TIME = 6 TO 12 MONTHS

FIGURE 11

Candidates - Voyager Experiments - Capsule

INSTRUMENT	WEIGHT (LBS.)	INSTRUMENT	WEIGHT (LBS.)
PHYSICAL CHARACTERISTICS OF ATMOSPHERE (MANOMETER, ETC.)	5	X-RAY DIFFRACTOMETER	10
MASS SPECTROMETER & GAS CHROMATOGRAPH FOR ATMOSPHERE	18	NEUTRON ALALYSIS	15
		MASS SPECTROMETER FOR SURFACE	20
SINGLE COMPOSITION DETECTORS	6	FLUORESCENCE SPECTROMETER	25
PARTICLE SPECTRUM ANALYZER	18	UV SPECTROPHOTOMETER	15
STEREO TELEVISION	80	J-BAND SPECTROPHOTOMETER	10
TOUCHDOWN DYNAMICS	26	OPTICAL ROTATORY DISPERSION	5
PENETROMETER	20	MASS SPECTROMETER & GAS CHROMATOGRAPH FOR BIOLOGY	20
PHYSICAL CHARACTERISTICS OF SURFACE (RESISTIVITY PROBE, ETC.)	14	FLUORIMETER	20
SEISMOMETERS (ACTIVE & PASSIVE)	99	MICRO CALORIMETER	5
GAMMA RAY SPECTROMETER	8	IR SPECTROMETER	15
PETROLOGICAL & BIOLOGICAL MICROSCOPE	15	PROBES (PHOTOMETRIC, BIOCHEMICAL ETC.)	24
		LUCIFERASE REACTION	10
			503
		PLUS AUTOMATED BIOLOGICAL LABS	200-850

TOTAL DATA = 10^{8} BITS
LIFE TIME = 6 TO 12 MONTHS

FIGURE 12

Candidates - Voyager Experiments - Orbiter

N66 31497

A FUTURE AUTOMATED BIOLOGICAL LABORATORY

By

Vance I. Oyama

NASA - Ames Research Center

INTRODUCTION

The accelerated advancements in space technology exemplified by each successful mission, make a soft landing on Mars within the next decade a real possibility. We now come face to face with implementing the task which the National Academy of Science has confirmed as our objective.

"*The biological exploration of Mars is a scientific undertaking of the greatest validity and significance. Its realization will be a milestone in the history of human achievement. Its importance and consequences for biology justify the highest priority among all objectives in space science--indeed, in the space program as a whole.*" [1]

With this charge, it is worthy of our deepest concern that a mission of this magnitude, cost, and man hours succeed in the specific tasks outlined:

a. "Determination of the physical and chemical conditions of the Martian surface;
b. determination of whether or not life is or has been present on Mars;
c. characterization of that life, if present; and
d. investigation of the pattern of chemical evolution, in the absence of life."

It is the purpose of this paper to consider research approaches to a meaningful automated biological laboratory to serve in the unmanned search for extraterrestrial life. In the process of elucidating those measurements and assays most critical and pertinent to life detection, it is becoming apparent that models of probable chemical evolution, and biochemical degradation in geological time, are required for the proper interpretation of any chemical composition found in any sample of extraterrestrial origin.

THE HISTORICAL FACT - LIFE ON EARTH

The historical fact - is that life exists on Earth. From this observational fact - the scientific approach is to extrapolate a hypethesis that biology, basically similar to that on the Earth is universal, and may exist in environments where carbon chemistry can prevail.

Among the planets in our solar system, Mars most nearly resembles the Earth in many environmental aspects, and it is not altogether impossible that Mars can support adapted terrestrial forms of life. Young, et al[2], Hawrylewicz, et al[3], and Scher, et al[4], have shown that many terrestrial organisms can tolerate some simulated martian environments. Considering the similarities of the two planetary environments, we would tend to accept carbon based chemistry as the most likely basis for martian life, if life exists. There is a possibility that martian life, though carbon based, may have evolved along different paths, perhaps utilizing a different set of amino acids, bases etc. It is difficult, however, to visualize life based upon an element other than carbon, as none such has been demonstrated to exist on Earth.

The manifestations of this carbon based life we know are distinctly different from those of the non-living matrix. Compositionally, all terrestrial life contains genetic, structural, catalyzing, membranous, lyophobic materials intimately

organized for a multiplicity of functions. The components of these organized units are characterized as nucleic acid, protein, carbohydrate, and lipid or conjugates of same, and appear without exception in every known organism, although minor variations in quality or quantity appear quite frequently. Nature has provided a highly efficient poised system arranged in discreet organized intracellular structures bounded by double membranes containing ribosomes and polysomes, nuclei structures etc.

To provide for such a dynamic chemical complex to exist and sustain itself, energy is constantly required. The energy-rich carbon molecules contain either oxygen-phosphorous, sulfur-phosphorous, or nitrogen-phosphorous bonds, with ATP established in all organisms as the common energy sink. Energy is transmitted by electron transfer systems mediated by enzymes. The enzymes, conjugate protein complexes, provide the catalysis for the variety of chemical reactions that must be carried on by the cell.

Now that it has been demonstrated that primeval atmospheres and lithospheres result in synthesis of biologically important compounds, we must try to make a distinction between the chemistries of non-living and living matter.

ABIOGENESIS-CHEMICAL EVOLUTION-THE DISTINCTION BETWEEN THE CHEMISTRIES OF NON-LIVING AND LIVING MATTER

The wide and variable distribution of organic compounds in recent sediments is primarily attributable to biological processes, though the organics in the eldest sedimentary rocks may not have this distinctive origin. Since the organics presently on the Earth are under the constant influence of biological turnover, thermal, hydrolytic oxidatial and radiative degradation, we cannot as yet describe from geochemical analyses the probable course of a biogenesis. We must deduce these processes from cosmic evidence and experimentally derive the chemicals produced under these "prebiotic" environments. The model composition assumed for the primitive Earth's atmosphere is proportional to the cosmic elemental abundances of carbon, hydrogen, nitrogen, and oxygen, which are in the form of ammonia, hydrogen gas, methane and water.[5] Starting from this primitive atmosphere, an impressive array of biologically important organics have been generated when a variety of energy sources have been applied (Table 1). It would be hazardous to claim, therefore, that an assortment of biochemicals alone would be indicative of life.

What types of analyses and how may we be enlightened from a knowledge of chemical composition? A preliminary estimation of total organic carbon will be necessary to determine the sample size required for specific organic analyses to follow. Analyses for soluble inorganic ions are necessary for a number of reasons: It may explain why precipitation had occurred in solution, e.g., ferri ion with hydroxyl, calcium ion with phosphate, etc..., why optical density occurs, e.g., the presence of auric, nickle, cobolt complexes, etc., why appareat toxic reactions occurs, e.g., the presence of high concentrations of selenium, arsenic, etc.. It may explain why biology is peculiar if discovered or non-existent. It will tell us whether extensive leaching occurred on Mars and the probable ionic requirement of a medium for culture, as organisms do have variable salt requirements. The available water content of the subsurface and surface may well be the limiting factor for the presence of life. Redox measurements will indicate the reducing potential of the environment and provide information on the energy state of the environment for culturing life, and hydrogen ion analysis will provide us with information on the availability of cations when coupled to the elemental analyses. In an actively metabolizing system changes in both redox potential and pH have been used as indicators of activity. Metastable molecules such as oxalacetic acid, pyruvic acid, citric acid, to name but a few, are maintained in dynamic equilibrium in biological systems and the expenditures of these molecules are counterbalanced by synthetic activities of the organism. Thus, even the existence of molecules of unstable natures would be of interest. Though the purines and pyrimidines found in the genetic materials are also found in abiogenetic systems, the lack of these bases may preclude at least the existence of terrestrial type life. The genetic material, proteins, polysaccharides, etc., of living systems are macromolecular in size. Most of the mass exclusive of solvent water is composed of polymers. Methods for the analysis of macromolecules become of importance to life detection, though the lack of polymers alone may be sufficient to rule out terrestrial type life, as in the case of the bases, their presence cannot be meaningful interpreted unless additional data could provide clues to the compositional and structural natures of these materials. If life did exist on Mars, the fossil record may contain polymers of interest, as polymers are more

stable generally than the monomers which make up their compositions. The prevalence of humic and fulvic acids in soils, acid hydrolysable proteinaceous residues in fossil shell material[15] all attest to the resistance of polymers in the natural environment.

The polymeric materials of life are sterically oriented and contain sterically oriented monomeric species which retain their individuality. The sterioisomerism represented by the monomeric units is optical isomerism, i.e., an isomerism which is characterized by the physical property of rotating plane polarized light. Polymeric biological substances such as proteins and nucleic acids, contain respectively, amino acid monomers and deoxyribose. The former contain a carbon element providing an asymmetric center whose four possible covalent linkages are satisfied by four different substituent groups. Likewise there are two such asymmetric centers available on deoxyribose. Both asymmetric amino acid and deoxyribose carbon elements integrally are part of the backbone structure of the chain, and contribute directly to the stereo-configuration of the overall macromolecular structure and conformation. The conformation is essential to life processes.

For the amino acids and the proteins which are composed of amino acid residues, terrestrial life without any known deviation has selected the "L" form in contradistinction to the "D" form for the saccharide moities. The structural and functional proteins are synthesized de novo from available amino acids provided from its nutrient medium or the cell synthesizes the "L" amino acids which are later incorporated. Thus the organism demonstrates a stereospecificity in synthesis. Protein enzymes have been demonstrated to selectively act mainly upon a particular isemeric substance (Table 2) "D" amino acid oxidases almost exclusively act to breakdown "D" amino acids.

In the simulated primeval systems abiogenesis results in the formation of racemic mixtures of amino acids, i.e., equal proportions of both "D" and "L" amino acids. Heretofore, no one has demonstrated asymmetric symthesis in systems simulating primeval systems. Though let us now examine the possibility that selection could have occured prior to the origin of life, and examine the implications of the data in respect to determining the significance of finding amino acids in macromolecular structures in its isomeric forms.

Disregarding the sequence of amino acids in polypeptides or proteins, and assuming that racemic mixtures of free amino acids were formed, we can estimate the probability of the spontaneous formation of isotactic molecules, i.e., molecules containing only one stereoconfiguration for the amino acids which make it up.

$$P = 1/2^n$$

where n = no. of amino acid residues in polymer of avg. molecular weight of 30,000 and avg. monomer mol. weight of 150.

$$P = 1/2^{200} \sim 1/10^{60}$$

Thus, it seems virtually impossible to assume that the stereospecificity of macromolecules could arise spontaneously by chance alone from a racemate. There either must have been some way in which stereo selection arose on Earth prior to the polymerization process, or the process of polymerization itself may favor or select the sequential addition of each succeeding member.

Enrichment processes by which stereoisomeric concentration could be attained have been suggested by a number of people[19,20]. Stereospecificity of quartz crystals upon the surface of which selective crystallization could occur, circularly polarized light promoting one isomeric form over the other in abiogenesis, diamagnetic influence of the Earth's field, and selective crystallization from saturated solutions, have been advanced and in the latter case demonstrated in the laboratory [21,22]. Harada has been able to demonstrate the selective crystallization from separate saturated solutions of racemic asparagine, aspartic acid, glutamine, and glutamic acid, by seeding with one of the isomeric amino acids in the presence of a high concentration of ammonium formate. But in the later system it still requires a seed crystal to start the process.

That the process of polymerization could influence the sequential addition of stereoisomeric species from a racemate has been investigated. Wald[23] on the basis of observations of Doty, et al,[24] suggests that the atactic peptide molecules, i.e., molecules containing mixed "D" and "L" configurations, are less stable thermodynamically, and, therefore, are synthesized at a slower rate than either isotactic molecular forms. But the validity of these observations could be questioned since the starting materials were not the amino acid monomers, but their carboxyanhydrides which are reacted in non-aqueous solvent systems.

Nevertheless, Doty's work has implications on the probable mechanism for asymmetric preponderance or selection. It is based on the fact that a minimum helix could begin the selective process. Eight amino acid residues are required for the minimum helix. The probability of synthesis of an eight membered isotactic polypeptide is:

$$P = 1/2^8 \sim 1/256$$

a much more probable occurence than the $1/10^{60}$ for the 200 amino acid configuration. Though isotactic molecules could prevail individually in the brew of the prebiotic environment, the individual amino acid residues taken in their totality would still be racemic. Thus, life in its propensity for "one sidedness" is required for the final enrichment process. Though we cannot preclude the possibility that two forms of life based upon stereoisomeric considerations might still exist upon a planetary body.

Since we cannot exclude the possibility that Doty type selective isotactic molecules could aggregate in isolated areas of Mars without biological intercession, the finding of isomeric enrichment in any particular sample cannot be used as proof positive that life exists or had existed, but the finding of racemic amino acids would give credence to abiogenic synthesis occuring on Mars, and provide strong presumptive evidence for the nonexistence of living matter. Figure 1 summerizes in block form the significance of stereoisomerism.

Under NASA sponsorship both Stanford University and the Ames Research Center are developing systems for the separation of stereoisomers[25-28]. In brief, the Stanford approach makes the separations by formation of diastereoisomers through a peptide linkage, whereas, the Ames approach utilizes esterification to provide the additional asymetric center. The precedent basis for the separtion of racemic amino acids was laid by Pasteur when he discovered that two of the three forms of tartaric acid would crystallize out separately from saturated solutions. Weygand separated diastereoisomers of peptides as their NTFA methyl ester derivatives[29]. Gil-Av and Narok demonstrated that racemic secondary alcohols could be resolved as the esters of optically active lactic acid derivatives[30], and Charles, Fischer and Gil-Av extended this to a few amino acids[31]. The Ames group have now been able to separate as many as fourteen of the most common naturally occuring amino acids as the NTFA sec-butyl esters by gas liquid capillary chromatography. Figure 2 shows the separation of a number of amino acids and the resolution of the isomers attainable with this method.

These approaches to the separation of compounds containing asymmetric centers apply to all substances which can be derivitized by a coupling agent containing an asymmetric center. It can be predicted that the O-silyl others or -methyl others of the uronic or gluconic acid enantiomorphs may similarly be separated with the secondary asymmetric alcohols. Additional significance may be attributed to this asymmetric analysis if "L" amino acid is coupled with D-sugar preponderance. This is the enrichment order accorded life on Earth.

METABOLISM

It is realized that, though chemistry and physical measurements provide answers on the environment as well as possible insight into potential biochemical pathways and the probably chemical constitution of life, these are not definitive. They do, however, provide data necessary for a more rigorous interpretation of measurements which have pertinence to physiological processes. One such physiological manifestation, metabolism, is essential to life processes. It may be described as the sum total of all chemical reactions which occur in an organism, the catabolic processes by which energy in chemical bonds are released by rupture and utilized for essential synthesis.

Chemical analysis can be usefully deployed to measure metabolic products generated by living systems. The evolution of $C^{14}O_2$ from C^{14} labeled organics has been demonstrated successfully on terrestrial microorganisms and soils[32]. The popular version has been dubbed "Gulliver". An absorber (getter) such as LiOH on a Mylar window scavenges carbon dioxide released from biological activity and the radioactivity is measured by a Geiger system. It may, however, be more pertinent to determine a larger assortment of gases rather than use a specific getter for each component of interest. This may be accomplished simply by providing an indiscriminate detector following a fractionating system such as a gas chromatographic column or mass spectrometer. Other chemical analysis to determine the synthetic products formed may also be considered. These analyses must provide a wide band analytical capability, i.e., a capability to assay a large spectrum of compounds, e.g., rather than a determination specifically for glucose, determinations for all known hexoses, pentoses, etc. It behooves us to seek general manifestations of metabolism which do not

assume a particular product to be formed and which is a ubiquitous manifestation of metabolism.

All organisms, be they photosynthetic, chemoautotrophic, heterotrophic, aerobic, contain molecules that are intricately concerned with either the transduction of radiant energy as in the photosynthetic organisms or in the transfer of electrons. Invariably, a tetrapyrrole structure linked by methylene bridges in involved. The hemes, cytochromes, and chlorophylls represent the types of entities containing this ring. When the cell dies, these structures of the respiratory activity of life, though modified gradually, resist natural degradation over long periods of time and are even found in sedimentary materials dated as early as the Upper Precambrian period[33]. Methods for the extraction, separation, and characterization of these residues are being considered for application. It must be emphasized, that it may be presumptive to assume from a static measurement a dynamic metabolic inference.

In order for the normal physiological processes of repair, growth, and reproduction to occur in biological systems, energy must be provided for these functions. Useful energy must ultimately come from the Sun in order to meet the energy balance, if life is to be maintained on Mars. The solar flux may be utilized and converted to chemical bond energies by non-biological conversion of chemical substances raised to energetic levels or it may be utilized by biophotosynthetic systems developed in the evolutionary path.

In non-photosynthetic organisms and in photosynthetic organisms in the dark phase, energy is derivable from energy in chemical bonds. The catabolic processes generate energy and chemical products of lesser potential energy. In exergonic chemical reactions, not all the energy generated can be utilized for work or synthesis. The energy not utilized is dissipated as heat. Since metabolism is basically a series of related chemical reactions, metabolic activity is always attended with heat output. To use heat loss as a measure of metabolic activity, it is necessary to differentiate this type of heat loss from those attending simple chemical reactions which do not involve biology. The differentiating characteristic common to all systems investigated is related to the biological population directly as long as the population is actively metabolizing. Thus, growth and reproduction--producing more mass and numbers--generate sustained caloric output in time, in contradistinction to simple chemical reactions which take place relatively rapidly, and are quickly dissipated when the concentration of the reactants become depleted.

Figure 3 shows results comparing optical density, viable count, dry mass and thermal output from a culture of E. Coli. The latter was measured with a Benzinger type microcalorimeter. Figure 4 shows a typical heat curve from a desert soil sample, and Table 3 summarizes some pertinent parameters and heat output characteristics.

GROWTH AND REPRODUCTION

Growth and reproduction are manifestations of living systems. Without growth, it is not possible to maintain reproduction indefinitely; without reproduction it is not possible for species to survive the normal ravages of radiation, freezing, drying, hydration, mechanical disruption, nutritive depletion, etc., over geological time. These attributes of life, growth and reproduction, are measurable by numerous techniques which depend upon changes in time. The changes invariably reflect a mass increase in material as distinguished from the medium proper. The measurement of light scattering changes is generally a reasonable and facile procedure in the laboratory. Other techniques, such as clonal plate counting, are generally used for counting, the viable population.

To extend growth detection techniques from a pure culture of known microbes to unknown materials, a number of factors must be taken into account. Factors such as osmotic shock, toxicity, nutritive balance, temperature, trace elements, etc., enter into the picture and may be directly responsible for the incompatibility between the organism and the medium. When nutritive materials, not physiologically balanced, surround the organism, the probability that the organism will survive the initial imbalance will depend upon a number of factors, such as the condition of the organism, the extent of the imbalance between medium and organism, the temperature, etc. The characteristics of growth curve lag period prior to exponential rise in numbers is ascribable to an adaptation response. A large inoculim is sometimes required when culture transfers are made in liquid medium if growth is to be obtained. The theory for this observation is that a large mass of living cells can tolerate the limited amount of imbalance and permit the cells to metabolize. Once the cells are capable of metabolizing, they can correct or compensate for the imbalance. Non-motile microorganisms adhere to surfaces, grow in soil pores, small crevices, and the like, which

limits the liquid to surface junctions as small as possible, but the growth and reproduction results in proliferation outward into the main body of fluid. It is possible, for example, to grow a single organism in liquid culture in capillary tubes, whereas it may not be possible if the cell is inoculated directly into a relatively large volume of medium.

In the light of the tenuous balance between the medium and the inoculum, a small amount of "soil" (10 mgs) in a relatively large volume of liquid medium (a few mls.) would not be the ideal technique for growing microorganisms. We not only reduce the size of the inoculum, but suddenly change the environment to which the organisms are accustomed. On the other hand, a relatively large sample of soil, e.g., (1 gram) in a few mls. adds (one hundred times) more inoculum and reduces the dilution of the soil particulates, and provides, therefore, more of the original environment to reduce the very probable imbalance. This technique will not permit light scattering measurements to be performed directly, as the background light scattering from the surfaces of the "soil" particulates will obviously interfere. It is possible, however, to use light scattering techniques with high soil to medium ratios.

One such technique applies a dilution scheme by which a heavy soil suspension is periodically diluted (Figure 5). If the rate of dilution is less than the rate of growth and reproduction, the "non-growing" soil particulates will be washed-out and growth will be measurable in optical density or light scattering units. The upper pairs of curves in both figures show optical densities in growing systems as differentiated from the lower pairs of curves in which no growth or subliminal growth is obtained. Since the particulate numbers for silt are greater than for sand on an equal weight basis, it is to be noted that greater numbers of dilution steps are required to reduce non-growing particulate signal.

Another technique in which particulate "soil" matter is spread upon a petri dish of semi-solid agar medium and scanned by a microdensitometer premits identifying particulates at zero time and at different incubation periods. Figure 6 shows particulate scan at zero and eighteen hours for a sterile soil sample, and Figure 7 shows clonal growth surrounding particulates after eighteen hours as compared to the zero time control for a non-sterile soil.

Another technique, conceived by Dr. Merek in our laboratories, is shown in Figure 8. A right angleplate of sintered glass supports the soil on the upper level. This support is in contact with medium in the lower chamber. The medium wets the soil by capillarity thru the sintered glass. Growth of microorganisms proceeds from the wetted soil, down the porous plate, and into the liquid broth. With this system no "soil" particles interfere with the photometric readings, the sample may be monitored for gases and the liquid and particulates generated assayed by other techniques. As in all growth measuring systems, it can never be assumed that a universally applicable medium can be devised. Therefore, as in all growth devices, a battery of different media must be provided. But, unlike the other techniques which utilize a small sample of "soil", a large sample of "soil" may be sufficient to provide all the necessary nutrients, and the addition of fluid is only essential to provide the vehicle for its solubility.

The generalization that the soil extract can provide the nutilites is tatamount to saying, "Wherever and whenever an indigenous mecrobial population exists, it exists only because the environment can provide all of the nutients required for its growth and multiplication."

It is obvious from the foregoing that there was an ommission of such general attributes of life as irritability, replicability, mutability, and perhaps sex. It is only because means for their determination are difficult to apply even in the laboratory. Other not so ubiquitous attributes such as motility, tropisms, phagocytosis, pycnosis, etc., have not been considered because of the degree of specificity implied.

The arguments for landing a series of replicate devices, each performing a single type of measurement may be weighed against a single site location of a multiple series of complimentary measurements.

The microorganism population may range from 10^9/ gram of surface soil in fertile agricultural soil to a few hundred /gram in desert soils. The variation in desert soils of limited rainfall is less than three orders of magnitude. A case in point is the distribution of trypticase soy broth viable organisms in the envirous of Death Valley, (Figure 9). If life is present on Mars, the expectation of wide variations on the surface is tempered by the generally inhospitable clime, the lack of water, low pressure, and high wind velocities. The averaging effect of the winds would broadcast widely any local differences ascribable to relatively circumscribed ecological environments.

Furthermore, it has been reiterated many times that no single type of measurement device will unequivocally demonstrate the existence of life[1]. In order to provide the controls for each measurement, complimentary measurements will be required especially on unknown samples.

AUTOMATED BIOLOGICAL LABORATORY

The complexities of a functionally integrated biological package which depends upon a number of complimentary measurements and assays upon a given sample or upon the products of a culture brew require for its simplification an effort to reduce the number of black boxes housing common electronic devices, detectors, and processing mechanisms, to the limits that reliability in redundancies would tolerate. When measurements as those described, require samples, it is obvious that an integrated sampling system makes sense.

The inadequacies of separate independent programmed systems for each device in terms of space allocation, power lines, and time sharing may be overcome by a computer-controlled system. It might be programmed to set the sequence of events, such that a chemical process which depends upon completion of a drying process ahead of it is not time dependent, but moisture dependent. When the moisture sensing probe's electrical signal reaches a threshold level, the next process is programmed in. In another mode it might reject a sample for other chemical steps because the amount of total organic carbon would be prohibitively small, but will still permit physiological tests to be performed. In like manner, if light scattering properties of a brew suggest a growing culture, the material could be transferred and chemical analyses performed on the particulate fraction. If a number of culture tubes show particulate increase, it will select the heaviest.

In such a laboratory, video systems can be usefully deployed to monitor reaction vessels, examine samples, or microscope images when other devices suggest important changes are to be observed. Within the laboratory, therefore, the video system becomes extremely useful as another complimentary detector.

This future automated biological laboratory will be a result of comparing numerous means of measuring life attributes, for their simplicity, sterilizability, unambiguity, instrumentability, sensitivity, robustness, stability, reliability, and dynamic range. From this initial survey, certain types of instruments, detectors, and related procedures will arise more prominently in their applicability than others. These may then be weighed on the basis of their essentiality, complimentarity, and integratability. From this, will certainly arise compromises in selecting the system and the components of that system. But it is hoped that the engineering considerations alone will not be the prevailing influence, but that - emphatically - a sound logic for life detection shall prevail.

SUMMARY

Though the expectation runs high that biologically important organics will be found on Mars, it does not necessarily follow that "life" exists on Mars. Chemistry, alone, therefore, may not suffice to establish the existence of life, but given complimentary physiological measurements such as metabolism, growth, and reproduction, the probability of unequivocal discovery becomes greater. A number of measurements or assay have been advanced which, when taken upon a given sample, will provide supporting necessary data. Strong inference that life does not exist may be made upon the positive finding of racemic amino acids. The physiological measurements become more meaningful if an array of tests are taken upon the sample. The efficient use of this laboratory depends upon a computer-controlled system. A future automated biological laboratory, therefore, is <u>one</u> experiment, logically conceived and implemented, to find the set of measurements which, taken together, will answer confidently-- "is there life on Mars"?

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CHEMICAL SUBSTANCES ABIOGENICALLY SYNTHESIZED FROM REDUCING ATMOSPHERES

BASES	SUGARS	AMINO ACIDS	TRIPEPTIDES
ADENINE	PENTOSES	GLYCINE	GLYCYLGLYCYLGLYCINE
GUANINE	DEOXYRIBOSE	ALANINE	GLYCYLLEUCYLGLYCINE
	RIBOSE	ISOLEUCINE	
NUCLEOSIDES	HEXOSES	LEUCINE	
ADENOSINE		ASPARTIC	**HYDROCARBONS**
DEOXYADENOSINE		GLUTAMIC	$C_1 \longrightarrow C_{46}$
		PROLINE	
NUCLEOTIDES		THREONINE	
ADENYLIC ACID		SERINE	
URIDYLIC ACID		LYSINE	
GUANYLIC ACID			
DINUCLEOTIDES			

TABLE I

SOME ENZYMES EXHIBITING STEREO SPECIFICITY

OXIDASES	PEPTIDASES
DECARBOXYLASE	DEHYDRASES
LIPASES	PHOSPHATASES
ESTERASES	DEHYDROGENASES
ACYLASES	ENOLASES
AMIDASES	PERMEASES

FROM: 1. A.H. BECKETT

2. G.N. COHEN & J. MONOD

TABLE 2

THERMOGRAM AND COUNT DATA: COMPARISON OF AMES SOIL SAMPLES

	TRYPTICASE SOY BROTH		SYNTHETIC STIMULANT MEDIUM	
	DV-1S	DV-3S	DV-1S	DV-3S
TIME TO FIRST MAJOR PEAK	———	48 hr 40 min	64 hr	155 hr
TIME TO SECOND MAJOR PEAK	73 hr 36 min	73 hr 36 min	118 hr	227 hr
TOTAL EXPERIMENTAL DURATION	110 hr	132 hr	159 hr	240 hr
HEAT OUTPUT AT FIRST MAJOR PEAK	———	540μ cal/sec	119μ cal/sec	117μ cal/sec
HEAT OUTPUT AT SECOND MAJOR PEAK	959μ cal/sec	466μ cal/sec	347μ cal/sec	89μ cal/sec
TOTAL INTEGRATED HEAT OUTPUT*	97 cal	93.3 cal	69 cal	42 cal
INITIAL COUNT (TOTAL IN CELL)	5.2×10^3	1.1×10^3	5.2×10^3	1.1×10^3
FINAL COUNT, END OF EXPERIMENT (TOTAL)	———	1.0×10^{10}	1.0×10^9	1.3×10^8

*TOTAL INTEGRATED HEAT OUTPUT ON DV-1S AND DV-3S WITH TRYPTICASE SOY BROTH WAS DETERMINED TO 100 hr, AT WHICH TIME, HEAT PRODUCTION HAD ASSUMED A CONSTANT VALUE FOR MORE THAN TEN hr. FOR DV-1S AND DV-3S IN THE SYNTHETIC MEDIUM INTEGRATION WAS PERFORMED FOR THE TOTAL DURATION OF THE EXPERIMENT: 159 AND 240 hr, RESPECTIVELY.

TABLE 3

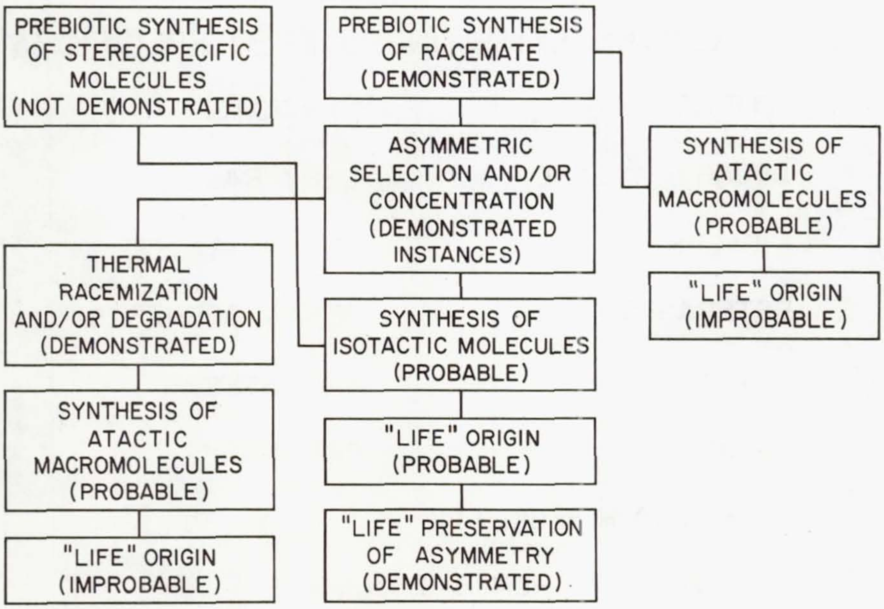

FIGURE 1

Stereoisomerism, an index to State of Evolution

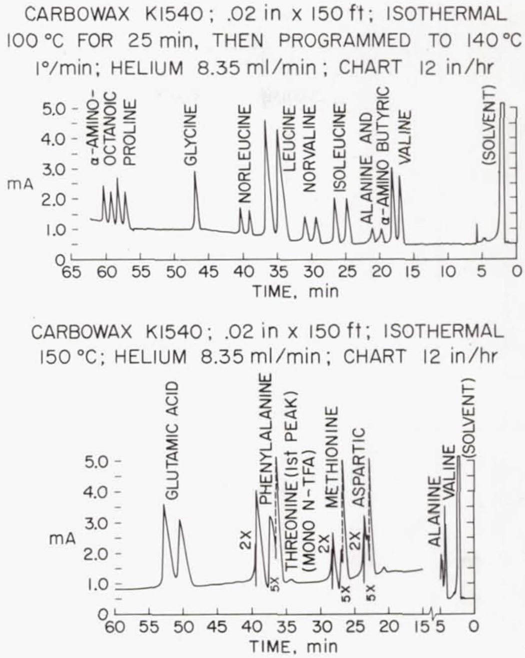

FIGURE 2

Separation of Racemic Amino Acids as Their N-TFA, Sec-Butyl-Ester Derivatives

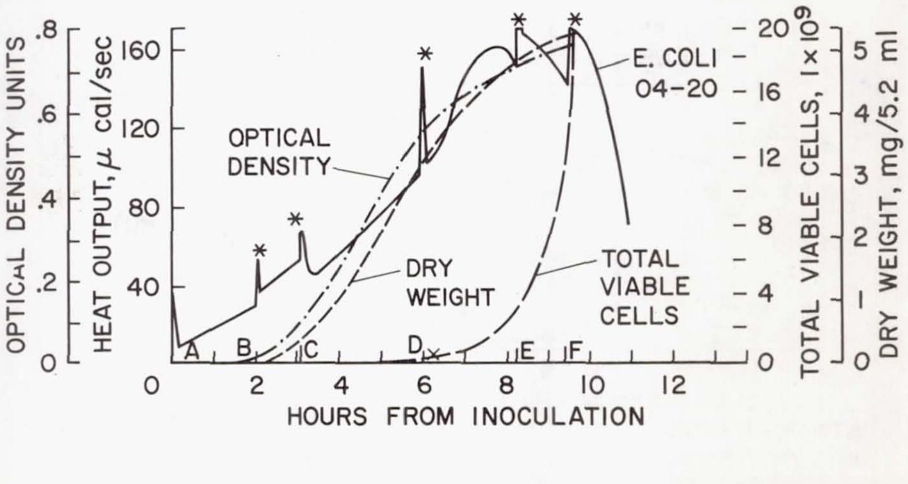

FIGURE 3

Comparison of E.Coli Thermogram to Mass, O. D., and Viable Count

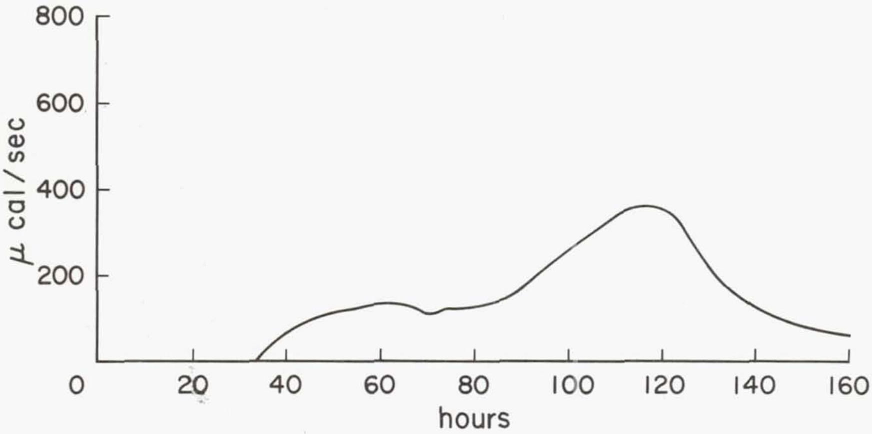

FIGURE 4

Thermogram of Ames Soil Sample DV-1S in Synthetic Medium

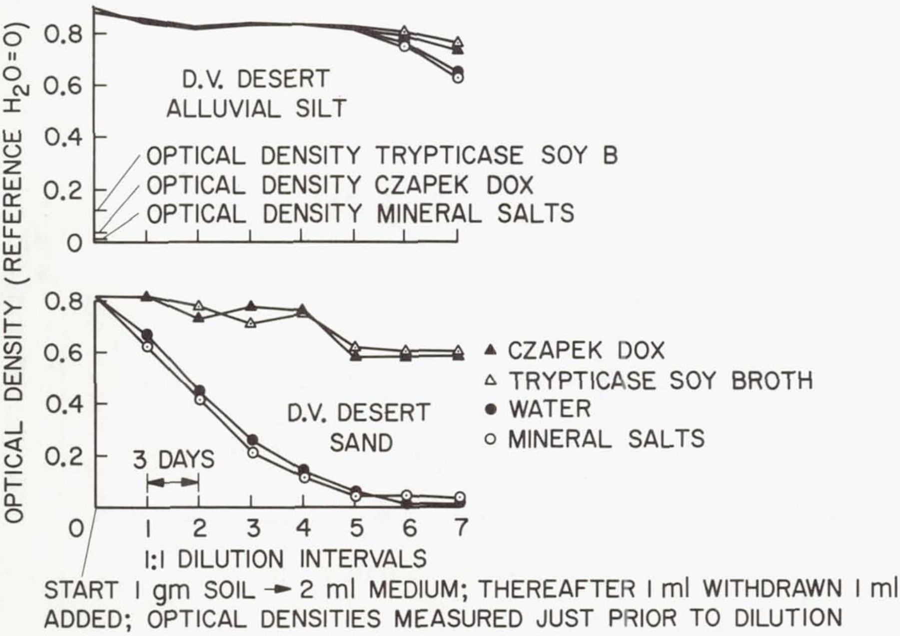

FIGURE 5

Growth Measurement-Dilution Concept on two Samples of Desert Soils

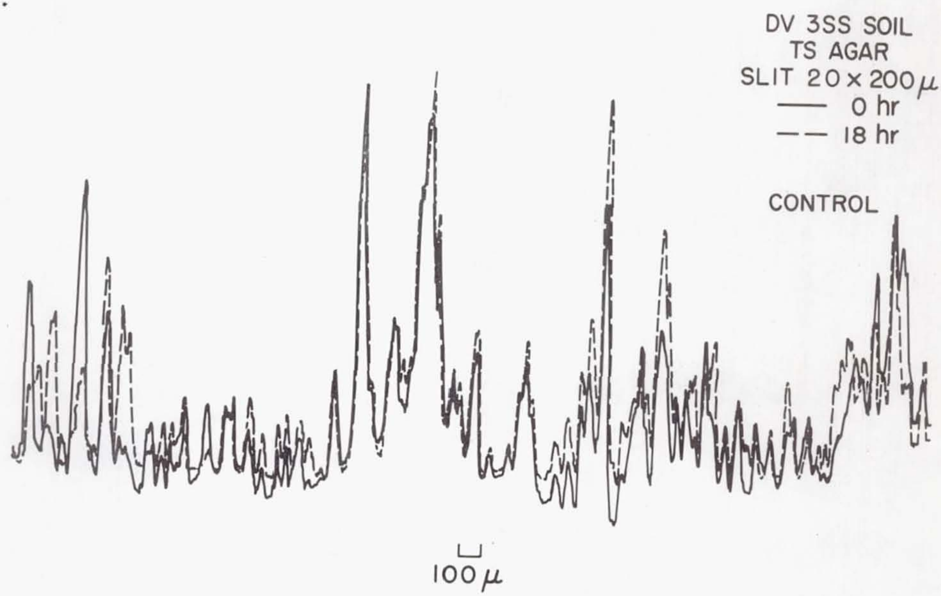

FIGURE 6

Microdensitrometric Scans of Sterile Soil Particulates Scattered Over Nutrient Agar

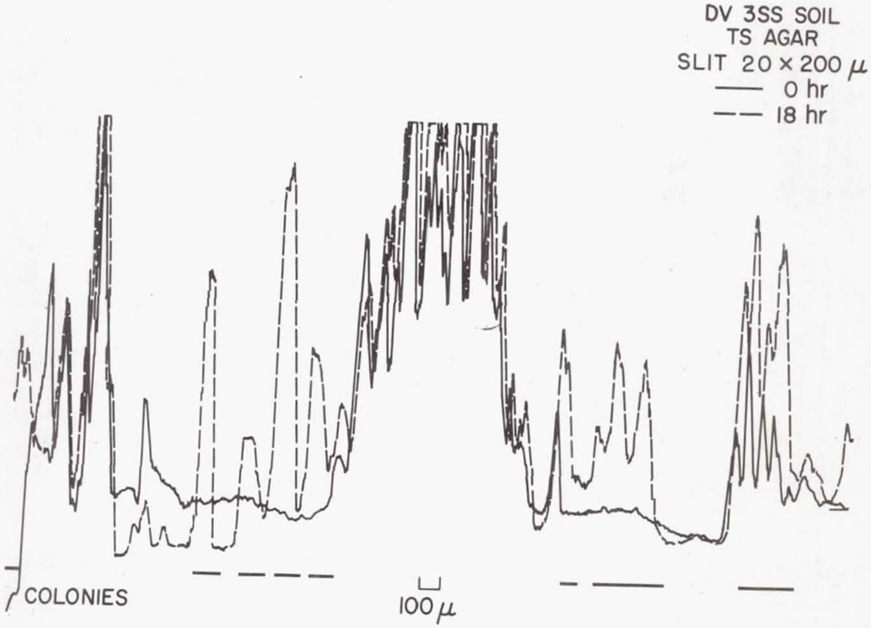

FIGURE 7

Microdensitrometric Scans of Native Soil Particulates Scattered Over Nutrient Agar

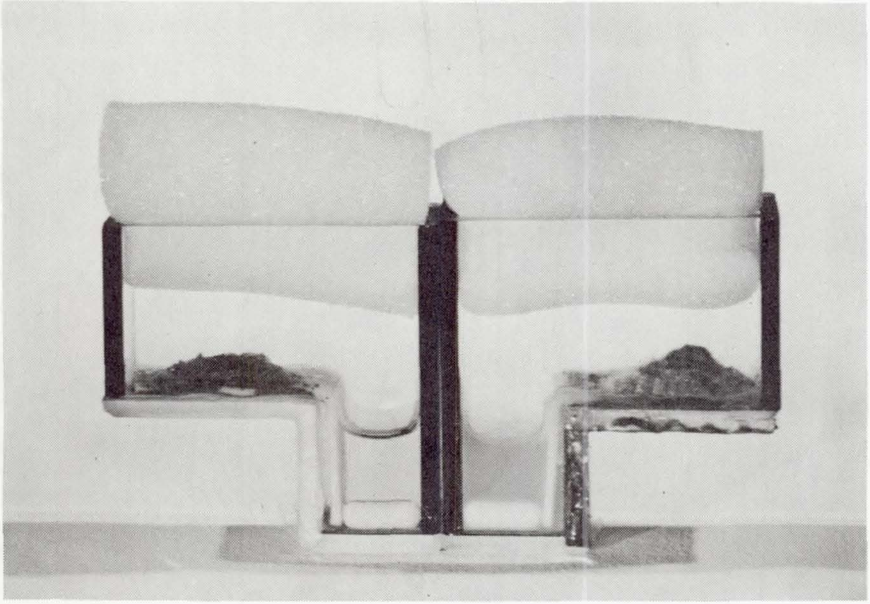

FIGURE 8

Technique for Measuring the Growth of Microorganisms

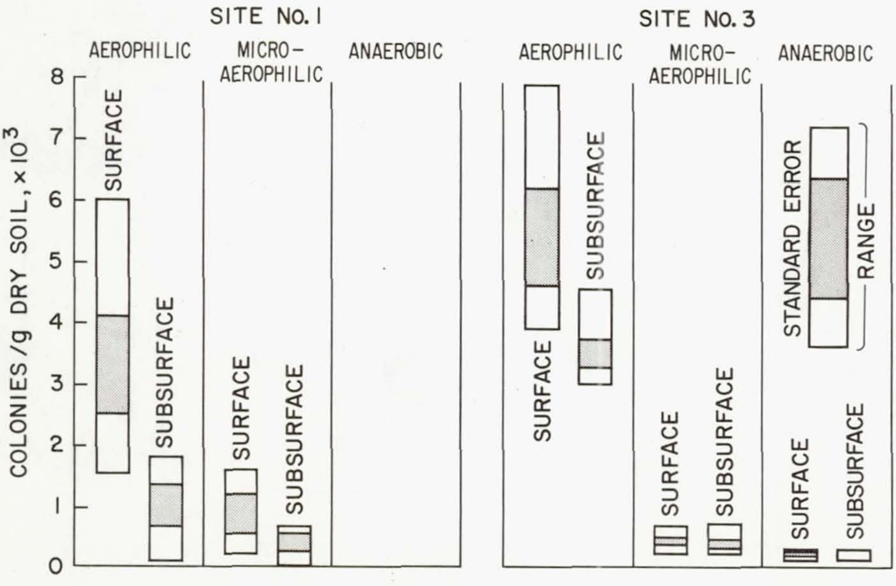

FIGURE 9

Colony Counts from Death Valley Sites

REVIEW OF MARINER 1962 AND 1964

SPACECRAFT DESIGN AND OPERATIONAL PLANNING

By

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INTRODUCTION

The technology of unmanned spacecraft is no more than about five years old. Since its inception, remarkable engineering progress has been made, and distinct classes of spacecraft have evolved. The Mariners represent one of these classes, having been designed to complete long-duration flights at great distances from the Earth.

Planetary missions place extreme performance demands upon the launch vehicle, spacecraft, and tracking stations. Venus and Mars reach positions favorable for conducting launches from Earth only every 18 months to two years, and they remain in these favorable positions for very brief periods of time. Consequently, a degree of technical and engineering conservatism must be invoked in planetary projects to ensure that the inflexible schedules can be met. Development or equipment delivery problems appearing late in a project could cause the cancellation of a mission and force a delay of up to two years. Launch planning to meet these very narrow and infrequent launch windows involved many unique problems.

Unprecedented distances are traveled in space during an interplanetary mission. Figure 1 shows the area of operations of the Mariner spacecraft. Mariner II journeyed 182 million miles for 3-1/2 months before arriving in the vicinity of Venus. This distance is roughly equivalent to 7,000 orbits of the Earth, or 700 trips to the Moon, and it is about one-half the distance to Mars. Mariner IV must operate for more than 7-1/2 months in order to complete its mission. The data collected by Mariner II near Venus had to be returned to Earth over a distance of 36 million miles, requiring more than three minutes to span the distance at the speed of light. The transit time for the data from Mars was more than triple this value, or approximately 12 minutes.

Long-duration deep space flights impose requirements for exceedingly high reliability, and operational plans for these flights must include many contingency provisions. The Mariner design philosophy embodies the concept of a spacecraft which is independent, self-sufficient, and 100 per cent reliable. However, complete implementation of such an approach is not wholly realizable within current weight and technological limitations; therefore, backup and midcourse command systems have been included in the design. This backup capability was needed during the Mariner II flight, for the secondary means of activating the Venus encounter mode had to be used. The Mariner IV spacecraft has been designed with additional redundant capabilities.

Although the subject of this paper is the Mariner spacecraft, the other essential elements of the total Mariner Project should be mentioned. A useful mission to Mars or to Venus would not be possible without the diverse talents and dedicated efforts of many organizations and individuals. The Mariner Projects have been built around four major system-related organizations--the Space Flight Operations System, and the Spacecraft System. Figure 2 is a pictorial representation of the equipment associated with these systems. Each of the major system organizations successfully completed extensive developments within its respective area to make the Mariner missions practicable. In addition, for the Mariner IV Project, over a thousand industrial contractors supplied equipment, and the scientific experiments aboard the spacecraft were provided by scientists representing eight universities.

In collecting the material for this paper, it quickly became apparent that time and space limitations would allow neither a detailed description of both Mariner II and Mariner IV nor a technical discussion of each of the closely interacting subsystems of the spacecraft. Thus the technical description is primarily a description of the basic characteristics of the Mariner spacecraft.

BASIC SPACECRAFT CHARACTERISTICS

All of the Mariners have carried a set of instruments designed to measure the fields, radiation, and particles of interplanetary space. There are about twice the number of nuclear particle detectors installed on Mariner IV as could be accommodated by Mariner II. The vector magnetic fidlds, measured by a flux-gate magnetometer on Mariner II, are detected by a more sensitive molecular resonance magnetometer on Mariner IV. The instrument for measuring the micrometeoroids of interplanetary space on Mariner IV has similarly been improved. The large quantity of very precise orbital data obtained during months of tracking the spacecraft away from the Earth has provided a basis for significant improvements in our knowledge of certain fundamental physical constants of the solar system.

Most of the Mariner interplanetary instruments and investigations are also useful in exploring the target planet. In addition, the spacecraft have carried special instrumentation for planetary measurements. Since Venus is enshrouded in clouds, and because of the payload weight limitations imposed on the Mariner II design, the planetary instruments on this spacecraft were limited to radiometers capable of spatially resolving the temperatures of Venus. The planetary instrument on Mariner IV is a television camera capable of taking about 20 pictures of the Martian surface. An extension in the application of precision tracking data obtained in the region of the planet should provide more accurate knowledge of the mass and atmosphere of Mars.

Figure 3 is a photograph of the Mariner II spacecraft. Figures 4 and 5 are of Mariner IV. Both utilize two celestial references for three-axis stabilization--for Mariner II, the Sun and the Earth and for Mariner IV, the Sun and Canopus. Both spacecraft also contain a midcourse guidance rocket, and they obtain their electrical energy from silicon cells. Mariner IV is larger and heavier than Mariner II. With solar panels extended and solar pressure vanes opened, it spans over 22 feet -- some six feet more than Mariner II. In the dimension from the top of the omnidirectional autenna to the bottom of the base structure, both spacecraft measure about 10 feet. Mariner IV weighs 575 pounds, approximately 125 pounds more than Mariner II, and contains almost 140,000 individual parts. It may be noted in Figures 3, 4 and 5 that the most obvious differences between the two spacecraft are in the number of solar panels, the shape of the base structure, the construction of the omnidirectional antenna masts, and the locations of the directional antennas and planetary instruments.

Because the Mars mission carries the spacecraft away from the Sun, additional solar cell area had to be provided for Mariner IV. Some 28,224 semiconductor cells are mounted on the four panels, and -- except during launch and maneuvers -- they provide the primary source of electrical power to the spacecraft. The attitude-control gas jets and the active solar vanes which stabilize the spacecraft relative to the direction of the Sun are mounted on the ends of the panels. It may be of interest to note that Mariner II was provided with a static solar vane to balance the pressure of the light from the Sun. The electrical energy originating in the Mariner IV solar panels flows into a pair of power regulators, either one of which can assume the full electrical load. Weight limitations precluded this redundancy in Mariner II.

The interior of the Mariner IV octagonal base structure contains most of the 35,000 electronic components, gas bottles, and regulators for the dual attitude-control gas system, the battery, and the propellant tank for the liquid-fuel midcourse motor. The midcourse rocket nozzle protrudes through one of the eight sides, and six sides are provided with polished metal thermal regulating louvers.

The 2,200-megacycle directional and omnidirectional antennas, most of the interplanetary scientific instruments, Sun sensors, and an insulating blanket are mounted on the side of the octagonal base which faces the Sun. The directional antenna differs from the one on an elliptic paraboloid weighing only 4.5 pounds, and is in a position fixed to point at Earth only during the latter half of the flight to Mars. The insulating blanket is made up of 30 layers of aluminized mylar sandwiched between a layer of teflon on the bottom and black dacron on top.

The side of the octagon away from the Sun provides the base for an additional thermal control blanket, the Mars acquisition sensors, the television camera, the cosmic ray detector, secondary Sun sensors, and the Canopus star tracker assembly.

The following are some of the more important engineering advances that were designed into the Mariner IV spacecraft:

1. Lighter-weight structures
2. Stabilization by star Canopus
3. Active solar vanes
4. Deep-space S-band communications
5. Restartable guidance rocket
6. Adaptive redundancy

Several of these innovations will be discussed in more detail later, but it should be mentioned here that most of the advances were required in order to make the more difficult Mars mission feasible, and all resulted from experience and work related to the earlier Mariner Projects. However, the last item, adaptive redundancy, requires further definition at this point.

The Mariner IV spacecraft includes two completely independent radio-frequency power amplifiers, two independent exciters for these radio amplifiers, two independent analog-to-digital converters, two independent oscillators for synchronizing the power frequency, two electrical power booster-regulators, and two completely independent pneumatic systems for stabilizing the spacecraft relative to the Sun and Canopus. While the selection of any combination of these alternate means of accomplishing various functions can be made by command from the Earth, another means of selection has been designed into the logic of the spacecraft itself. For example, if one of the radio frequency power amplifiers fails and a command cannot be sent, the spacecraft will, within a period of time, automatically switch to the other power amplifier.

One way of grossly assessing the design tradeoffs and determining how the technological innovations have influenced the spacecraft configuration is to compare the relative weights of the various subsystems. Table I shows these weight comparisons. The structural weight fraction of Mariner IV is substantially less than that of Mariner II. The mission requirement for additional solar-panel area to provide adequate power and redundant electrical booster-regulators is reflected in the weight increase in the electrical power subsystems. In this case, the capability was more than doubled, but the weight was increased by only about 10 per cent. The increased weight in the scientific instrument and data-handling subsystems is an indicator of the improvement that has been made in the data-gathering capacity of the spacecraft.

TABLE I

	Mariner II	Mariner IV
Structure	77.22	79.16
Antenna	19.78	7.43
Radio	19.25	34.40
Command	8.76	10.12
Power	62.11	71.02
Solar Panels	43.20	79.02
CC&S	11.21	11.38
Data Encoder	13.61	22.43
Data Storage	-----	16.89
A/C and Autopilot	53.26	63.24
Actuators and Pyro	9.07	12.21
Cabling	37.77	44.66
Propulsion	31.36	47.47
Thermal Control	7.17	15.43
Science	48.72	59.31
	442.49 lbs	574.17 lbs

It is profitable to review the launch of Mariner III. Although it was a failure due to the apparent collapse of the aerodynamic shroud, the lessons learned are worthy of review.

Liftoff occurred at 19:22:04 November 5, 1964 GMT. Approximately sixty minutes after launch and after analysis of engineering telemetry data, the Spacecraft Performance Analysis and Command team at JPL (SPAC) reported that the solar panels were not deployed and that the spacecraft was still on battery power. When it became apparent that there was no solar power and that the spacecraft attitude control system was not functioning properly, a decision was made to send a command to the spacecraft to turn off the gyros prior to the nominal time in order to conserve battery power.

Analysis of the telemetry indicated that the spacecraft was not attitude stabilized and the assumption was made that either the Agena had not separated or that the shroud was still surrounding the spacecraft. Lockheed calculated that the spacecraft separation rate indicated the shroud was still on during separation. The computation of C_3 (the injection energy) confirmed this fact. The Flight Path Analysis and Command team (FPAC) reported that the Agena Tracking data indicated that the Agena had indeed separated. The decision was made to attempt a midcourse maneuver in the hope that this might shake the spacecraft loose. However, the maneuver command was not executed since the battery was depleted before it could take place.

There was an operational lesson to be learned from this event.

The organization during the launch as regards the spacecraft is divided into two basic parts. One part is the launch team at Cape Kennedy, Florida. These are the engineers, technicians, and supervisory personnel that are responsible for bringing the spacecraft into being. It is our philosophy at JPL to retain the continuity of responsibility from inception to completion. In this case completion is at least through the successful launch. The other part is the Flight Operation Team which resides at JPL in California. These people are trained to analyze the data available through telemetry and to recommend action.

During the traumatic experience of Mariner III launch we suddenly found ourselves with the "best" brains at Cape Kennedy and the "best" operational capability at JPL. It is undoubtedly true that nothing could have salvaged Mariner III. Nevertheless, our future launch operations must take into account this divided capability. The result of this division was a suddenly divided responsibility and an accompanying confusion. In the future the prelaunch training program should take into account this division of talent and place the operational responsibility absolutely in the hands of the flight operations from the moment of launch.

Mariner IV was launched from Launch Complex 12 at the Air Force Eastern Test Range (Cape Kennedy) on November 1964, 14:22:01 GMT. For those readers who like statistics, I offer the following information. Launch occurred with an inertial azimuth of 91.4 degrees and performed a programmed pitch maneuver until booster cutoff. During sustainer and vernier stages, adjustments in vehicle attitude and engine cutoff times were commanded, as required, by the ground guidance computer to adjust the altitude and velocity of Atlas vernier engine cutoff.

At 14:27:21 GMT the shroud separated from the Agena followed by a hugh exhalation from the Earth-bound observers. After Atlas/Agena separation there was a short coast period prior to the first ignition of the Agena. At the preset velocity value the Agena shut down and went into a near circular parking orbit at an altitude of 188 kilometers and a speed of 7.80 kilometers per second.

After a period of 40.87 minutes in this parking orbit the Agena was restarted. Ninety-six seconds later the Agena was cut off. This occurred at 15:04:27 GMT over the Indian Ocean at a geocentric latitude of -26.25 degrees and a longitude of 68.82 degrees. At this time the Agena/spacecraft was at an altitude of 197.2 kilometers and traveling at a speed of 11.443 kilometers per second. At 15:07:08 GMT the spacecraft separated from the Agena and began its solitary sequence of events.

I will not pursue the details any further. The deployment of the panels and the acquisition of sun proceeded normally and sun acquisition was complete at 15:31:00 GMT.

The sequence to acquire Canopus was complete at 10:59:26 GMT on 30 November 1964. Approximately 3 hours after this event a transient occurred. The implication of this event was not felt until later. At 13:42:00 GMT a transient was observed in the roll channel. The spacecraft did not lose Canopus. At 10:00 GMT 2 December the roll axis dropped Canopus and the gyros came on. Canopus was immediately reacquired and the gyros turned off.

On 4 December the midcourse maneuver was attempted. The instructions were loaded into the spacecraft maneuver system via command. At 14:35:00 GMT the command to execute the maneuver (DC-27) was transmitted. At 14:38:00, or 3 minutes later, the spacecraft went into a roll search mode. This was nearly catastrophic since the maneuver had not yet been initiated by the spacecraft internal logic and the turns are referenced to the initial spacecraft attitude. Fortunately, our designers had the foresight to provide a maneuver inhibit command (DC-13). This was immediately transmitted and the spacecraft responded.

It is pertinent at this point to describe the situation at that time with respect to the roll control system. Figure 6 shows the predicted star map for the Canopus sensor. The star Canopus is at 0 or 360 degrees.

The logic of the Canopus acquisition was established to reject objects either two bright (such as Earth) or objects to dim to be Canopus. However, objects such as Gamma Velorum (G-Vel) are acceptable to the gate logic. Thus when the roll control was disturbed the system sought a new target and found it. However, we did not at this time understand why this inadvertent loss of Canopus lock.

On 5 December the turn instructions were reinserted and at 14:25 GMT the execute command was again transmitted. This time the maneuver proceeded flawlessly. The spacecraft disengaged itself from the Sun and Canopus. It pitched over -39.23°, rolled 156.08° and then ignited the motor for 20.06 seconds for a velocity increment of 16.7 meters/second. Figure 7 shows where the spacecraft would have arrived without the maneuver and then where it arrived with the maneuver.

With the midcourse maneuver successfully completed the Operations team settled into the routine of the long cruise and the preparation for planet encounter.

An Encounter Preparation Planning Group was established for the purpose of making the following studies:

1. Possible failure modes of the spacecraft which affect encounter.
2. Prepare logic diagrams of the spacecraft for determining failure modes and the necessary corrective action.
3. Determine the encounter conditions under the various failure modes.
4. Recommend any action that might enhance the encounter such as tests, sequence of commands, etc.
5. Determine the operational needs to accomplish the previous items such as hardware, computer programs, etc.

An early conclusion of this group was that the intermittent loss of Canopus was due to the presence of dust. This dust, presumably on the spacecraft at liftoff, when disturbed, drifted out to where the Canopus sensor would see it. For example, a speck of dust only .10 millimeters across and 800 meters from the spacecraft when illuminated by the Sun appears brighter than Canopus. This triggered the intensity gate logic, the tyros would come on, and the spacecraft would seek a new target.

It was decided to send a command that in essence inhibited the intensity gate action. This was done and the spacecraft never again lost Canopus, at least up to the date this paper was written. Many disturbances of the intensity have been noted. Some of these have been accompanied by roll attitude disturbances. However, our telemetry sampling technique does not allow for observance of simultaneous events. There does not appear to be a correlation between the intensity disturbances and cosmic dust activity. But this may be due to the gradual depletion of Earth-originated dust while the cosmic dust quantity varies as some function of the distance from the Earth.

The solar pressure is of sufficient force that the particles are swept out of the field of view more rapidly than the tracking capability of the spacecraft.

Our experience with the Canopus sensor raised concern over the nominal encounter sequence. At liftoff a cover protects the optics of the television system. It was planned that this cover would also protect the optics against cosmic dust abrasion during the long cruise, and just prior to encounter that a timed event would release the cover. A spring snaps the cover free, exposing the optics.

This snapping action could release a cloud of dust that might have confused the sensor at a critical moment. It was decided to drop the cover early while the telemetry could be received over the low gain (almost attitude independent) antenna in case we had an attitude control problem during the exercise.

On February 11, 1965 the scan cover was dropped by initiating the encounter sequency by command. This was demonstrated to have been a wise move when later analysis of the engineering data revealed a near loss of roll control.

Our analysis seemed to be confirmed by the docile performance of the spacecraft after the aforementioned exercises. The concern with the dust heightened as our cosmic dust detector showed a continuous increase of activity as we went out. It was remembered that Mars I (the unsuccessful Russian '62 launch) failed after achieving a record 66 million mile communications record. As we approached this distance the cosmic dust maintained its increasing activity. On April 29, 1965 Mariner IV exceeded the record established by Mars I of 66 million miles. About that time the dust activity started to subside.

Zond II (the Russians' 1964 Mars attempt) trailed Mariner IV by about one million miles. It was later reported that Zond II failed on May 2, 1965. Figure 8 shows an interesting correlation between Mars I failure, Zond II failure, and Mariner IV cosmic dust activity. This study was performed by P. Feitis at the Jet Propulsion Laboratory. It must remain in the realm of speculation as to whether or not that cosmic dust caused the failure. A non-communicating spacecraft is very difficult to analyze. I will leave the subject except to note that it would be very nice to know what we did right that they did wrong.

Our planning for encounter proceeded to examine the possible Martian environment and its potential effect upon the spacecraft performance. Two prime candidates for concern were the expectation of increased dust near the planet and the possibility of a "Van Allen Belt." The dust problem has been discussed before. The radiation problem was studied on the basis that a high level of energy might affect the electronic components. In particular, the Canopus sensor signal to noise ratio under intense radiation was studied.

Tests utilizing accelerators were implemented and all critical subsystems were exposed to levels of radiation sufficient to establish their tolerance. Our project scientists established a best estimate of the maximum radiation we might expect at Mars if a "Van Allen Belt" existed. Figure 9 shows the estimate and the results of our tests. We were therefore quite confident of our ability to tolerate the Martian radiation environment.

The planned sequence of events for encounter were examined and they revealed several weaknesses in our logic. Figure 10 shows graphic layout of the anticipated encounter.

Figure 11 shows the technique by which the scan platform acquires the planet and then tracks it. The jagged lines represent the scan platform action. For simplicity the time between scan cycles has been exaggerated. It takes 12 minutes for the platform to traverse the 180° sector. The platform scans until the planet enters the field of view (48.5 degrees of the Wide Angle Mars Gate (WAMG). The WAMG then operates the platform so that it tracks the planet. Another sensor called the Narrow Angle Mars Gate (NAMG with a field of view of 1.5 degrees senses the planet at the proper time and initiates the photographic sequence.

As with all our critical functions, there is redundancy. The backup to the WAMG was command DC-24. This command stops the platform. The fallacy of using DC-24 is that if it has been decided that the WAMG has failed, then DC-24 would be sent so as to stop the platform at 179.43 degrees. However, the spacecraft communication distance is about 135 x 10^{+6} miles. It takes a command at least 12 minutes to arrive. In this time another 180 degrees are traversed. If in this pass the WAMG acquires, then the DC-24 command will inhibit the platform at this angle at that moment. If this should happen prior to about $^1/_2$ hour before encounter, the platform would be positioned so as to miss the planet altogether.

To avoid this dilemma and to still maintain our philosophy the procedure was reversed. We sent the command to position the platform before it was possible for the planet to be in the field of view. It was planned to reinitiate the planet sequence and acquire with the WAMG if telemetry indicated that the DC-24 exercise had resulted in an unacceptable platform angle.

During the encounter this was highly successful and the platform stopped by DC-24 at 178.45 degrees which was well within our tolerance.

With the platform properly positioned our next concern was the initiation of the picture sequence. Our backup in case the NAMG failed was command DC-16 which also initiates the sequence. This was transmitted to arrive about five minutes after narrow angle acquisition. However, this proved to be unnecessary, since the narrow angle acquisition was excellent as noted by the first photograph, which clearly shows the limb of the planet, Figure 12.

The next critical event was the proper recording of the photographic data. Because of the limited communication bandwidth the data is recorded on a magnetic tape. The machine is of a continuous loop type. It does not reverse and rewind to prepare for playback. It is a characteristic of our recording technique that all previous data is erased by new data. There are two tracks on a 330 ft. loop of tape capable of storing 5.24 x 10^6 bits of data.

During the record cycle the vidicon data is read into the tape for a period of 24 seconds. The tape at this time is running at a speed of 12.84 inches/sec. It then takes another 24 seconds to erase the vidicon and to prepare for the next picture.

To conserve the tape the recorder is stopped between pictures. The nature of our concern is indicated in Figure 13. The normal sequence is noted at the top. The only telemetry signal available to us is the End of Tape (EOT) which signals the change to the second track. As can be seen by the lower sequence, if the stop circuitry fails, because of the 12 minute communication time, the system would have erased all of the data by the time the backup command arrived.

We therefore decided to send command DC-26 which terminates also the record sequence through a different method. This command was sent to arrive just after the dark limb.

Although we anticipated some sort of problem of this nature during encounter, and felt that our transmission of DC-26 would salvage some pictures, it still came as a shock when at five minutes after the photo sequence we noted an end of tape signal. Then about 9 minutes later the correct indication was received. The correct signal was also received at the end of the photo sequence when the system completed its 22 pictures and switched to mode 2 (cruise mode).

As yet we have no explanation for the false telemetry. Because of the presence of correct indications we were certain that the performance was correct and because of our DC-26 utilization we were confident that we had salvaged some pictures.

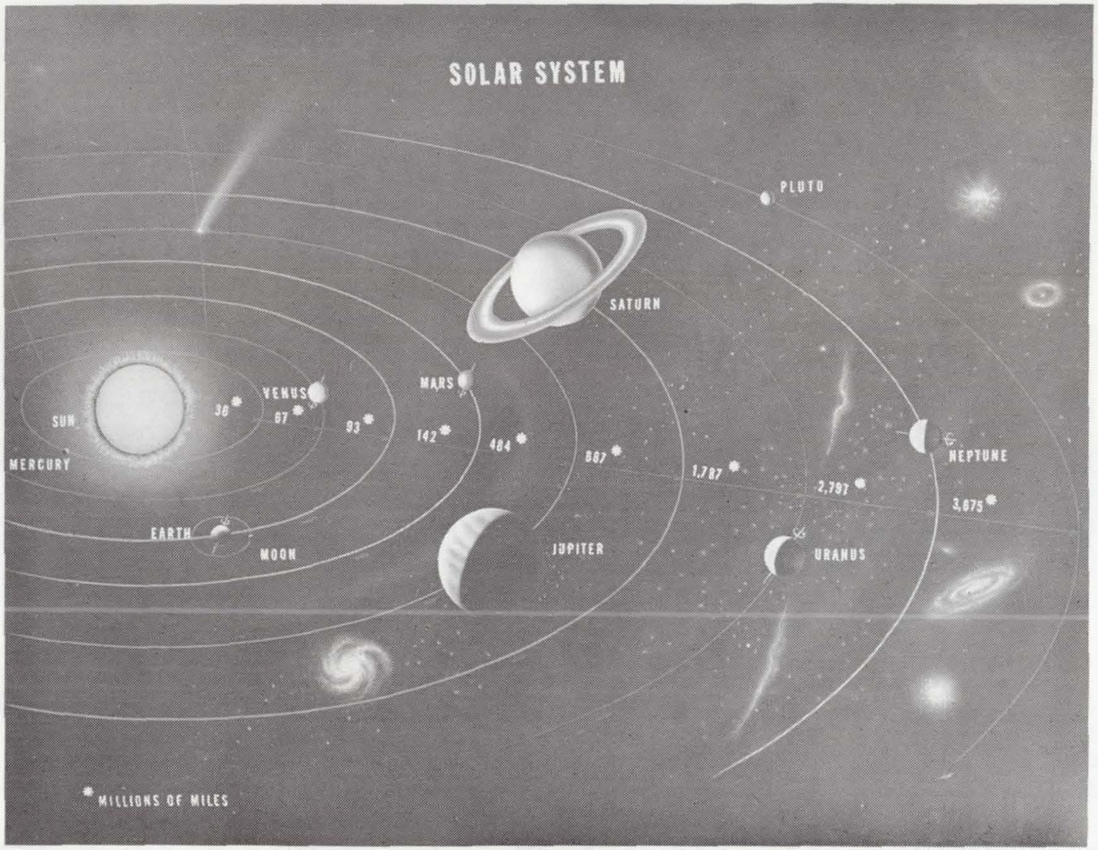
Utilization of DC-26 also turned off all of the scientific equipment. The command to restore the cruise mode (DC-2) was sent one minute later. Thus there was a modicum of concern about the restoration since this was the first time since midcourse that the cruise science had been turned off. Again, performance was excellent.

Mariner IV, at the time this paper is presented (August 27, 1965), is 271,906,422 km from the Earth and 17,041,582 km from the planet Mars. It is traveling at a velocity relative to the sun of 77,726 km/hr. Prior to encounter it was inclined to the ecliptic 0.129 degrees. Its present inclination is 2.54 degrees. This is due to the effects of the gravitational pull of Mars during the flyby. On September 1, 1967, the spacecraft will reach aphelion at a distance of 235 x 10^6 km from the sun. It will remain in orbit about the sun with a period of 587 days. On June 5, 1966, it will be at perihelion, 165 x 10^6 km from the sun. Thus it remains outside the orbit of the Earth.

Its maximum range relative to the earth will be 348 x 10^6 km. We anticipate maintaining radio contact on a periodic basis until we are convinced that the spacecraft has failed. We will utilize the 210 foot antenna at Goldstone, California for this purpose. Unfortunately, the extreme range precludes telemetry.

The spacecraft will be nearly occulted by the Sun about April 1, 1966 at a distance of about 330 x 10^6 km. The spacecraft will appear about $^1/_2$ a degree above the sun. Communication at this time may be impossible because of the noise of the sun and the possibility of focussing the Sun's heat onto the antenna and melting it.

On September 8, 1967, the spacecraft will be at a distance of 46.9 x 10^6 km from the Earth. Just prior to this time and for a period following we will be able to receive telemetry from the spacecraft over the low gain antenna. As can be seen in Figure 14, we are able to receive telemetry independent of the roll attitude of the spacecraft. This is also a period of time of interest in the solar cycle, Figure 15. On the pessimistic side is the fact that the spacecraft will have been in space for 3-$^1/_2$ years. It will have traveled 1,373,242,000 miles. We are presently planning this operation. Our confidence in Mariner IV is high and although the return in '67 is optimism of the highest order, the machine has yet to let us down.



SOLAR SYSTEM
PLUTO
SATURN
SUN
MERCURY
VENUS
MARS
36 *
67 *
93 *
142 *
484 *
887 *
1,787 *
2,797 *
NEPTUNE
3,675 *
EARTH
MOON
JUPITER
URANUS
* MILLIONS OF MILES

FIGURE 1

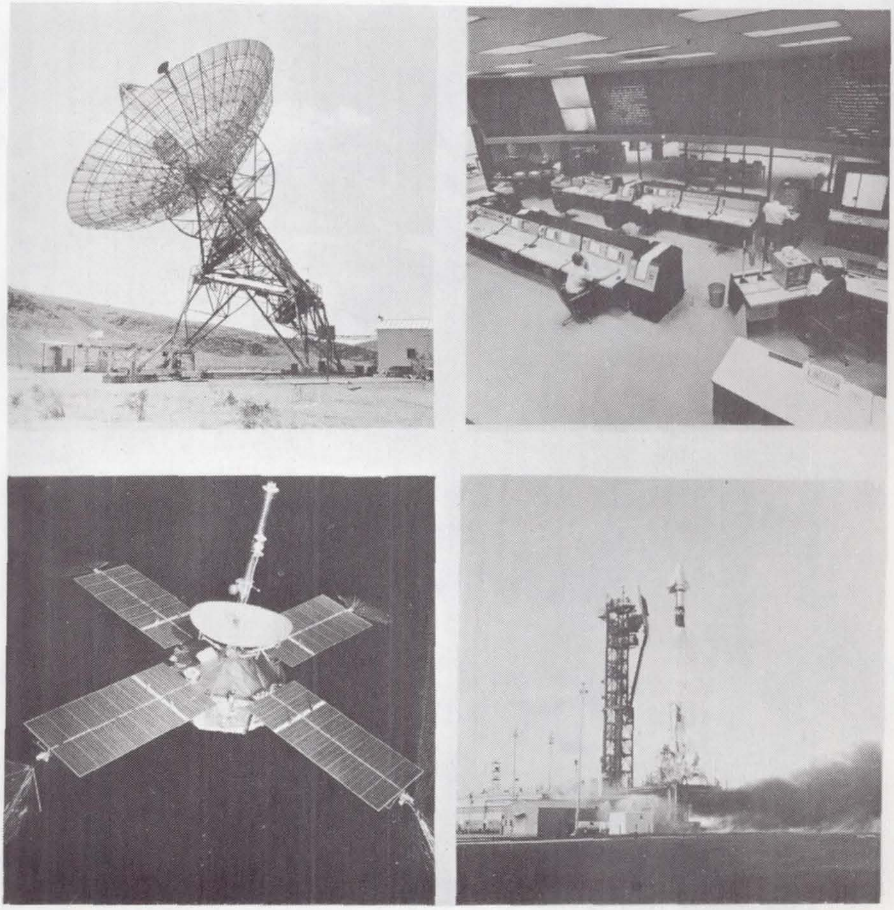


FIGURE 2

Space Mission System

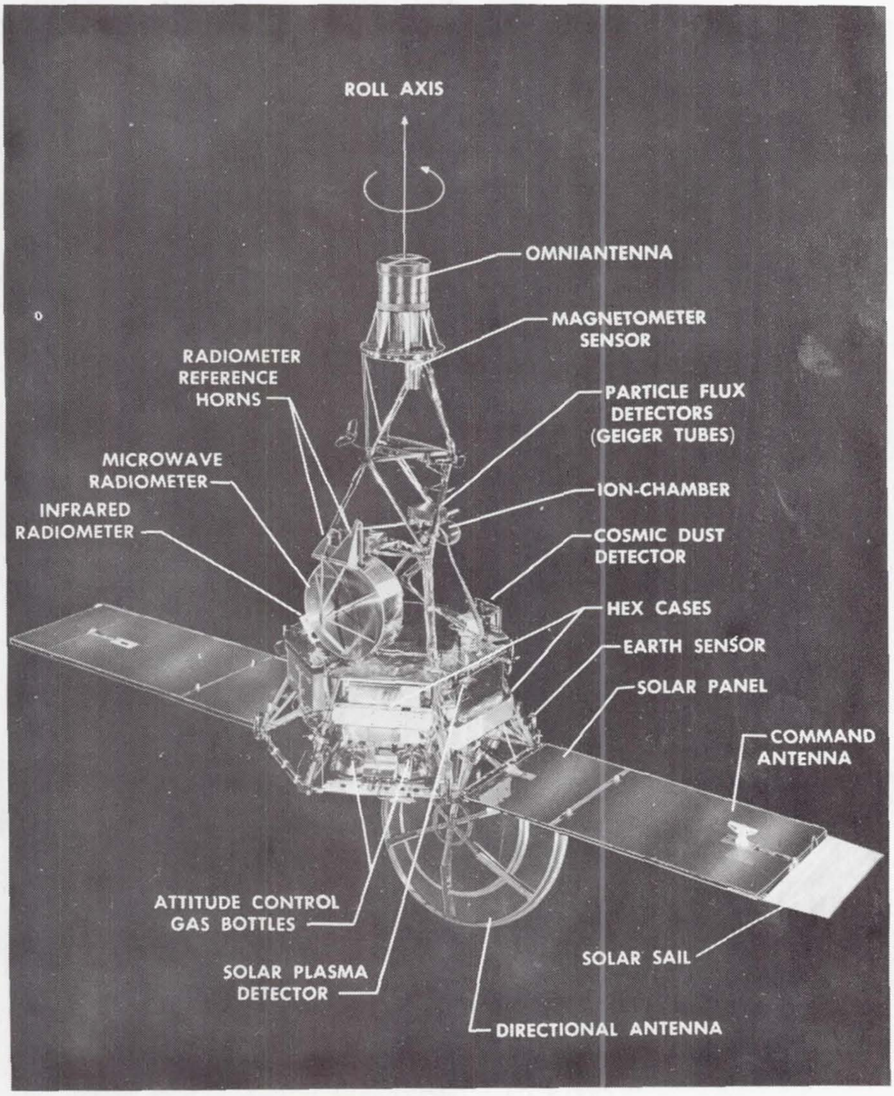


FIGURE 3

Mariner II Configuration

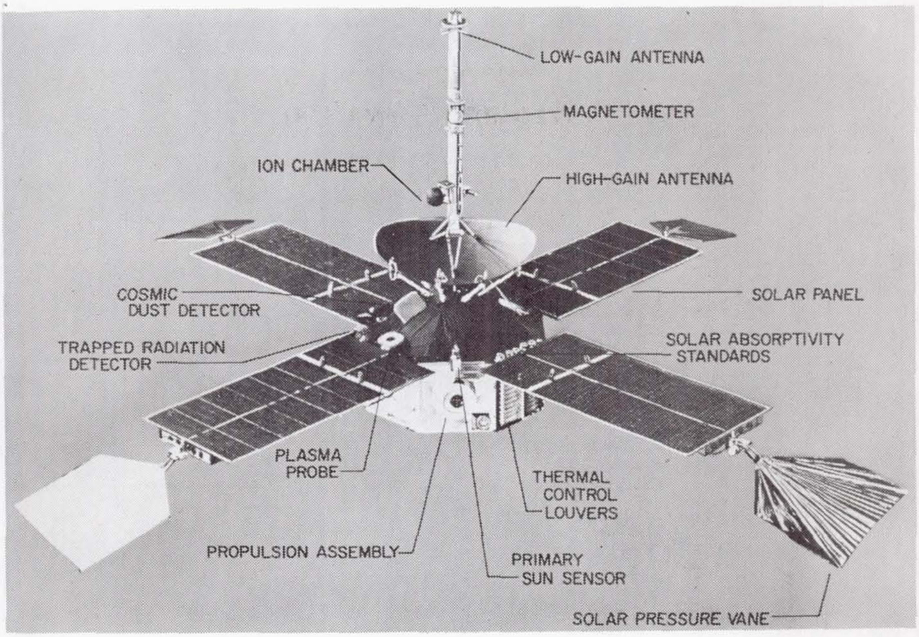


FIGURE 4

Mariner IV (Top View)

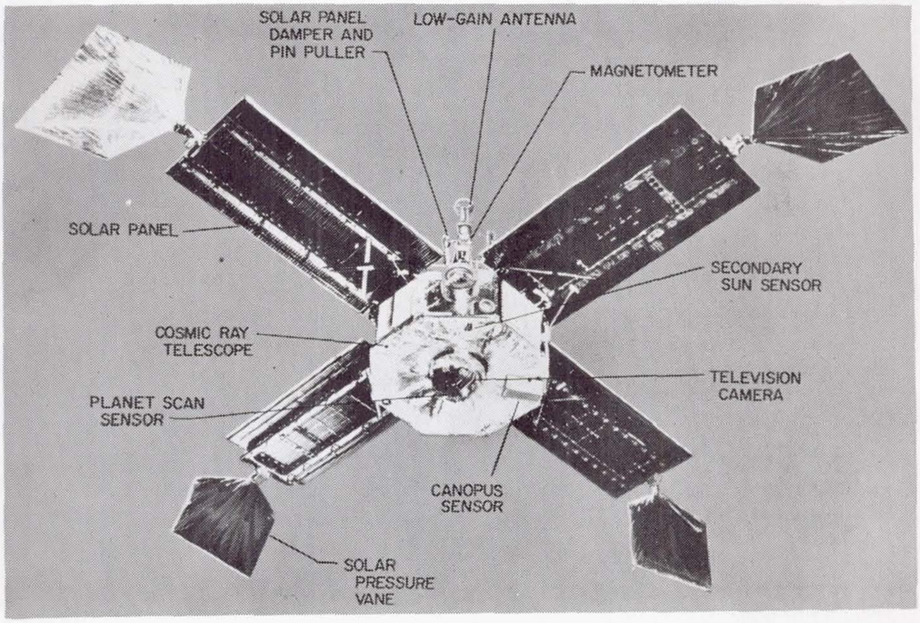


FIGURE 5

Mariner IV (Bottom View)

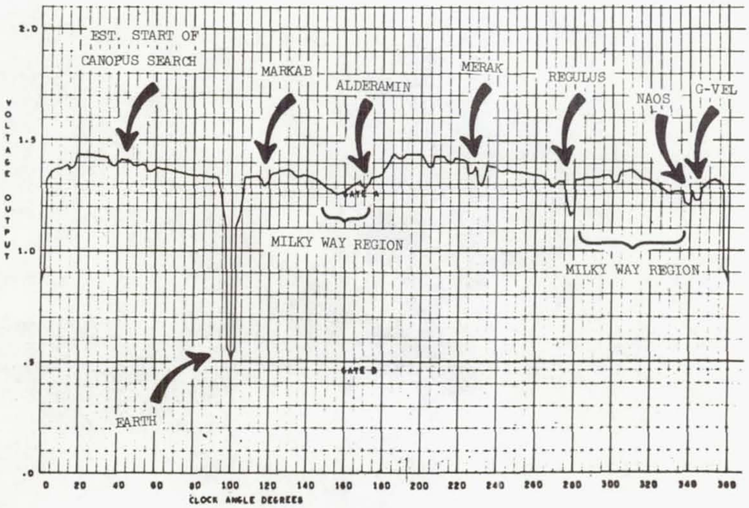


FIGURE 6

Canopus Star Map

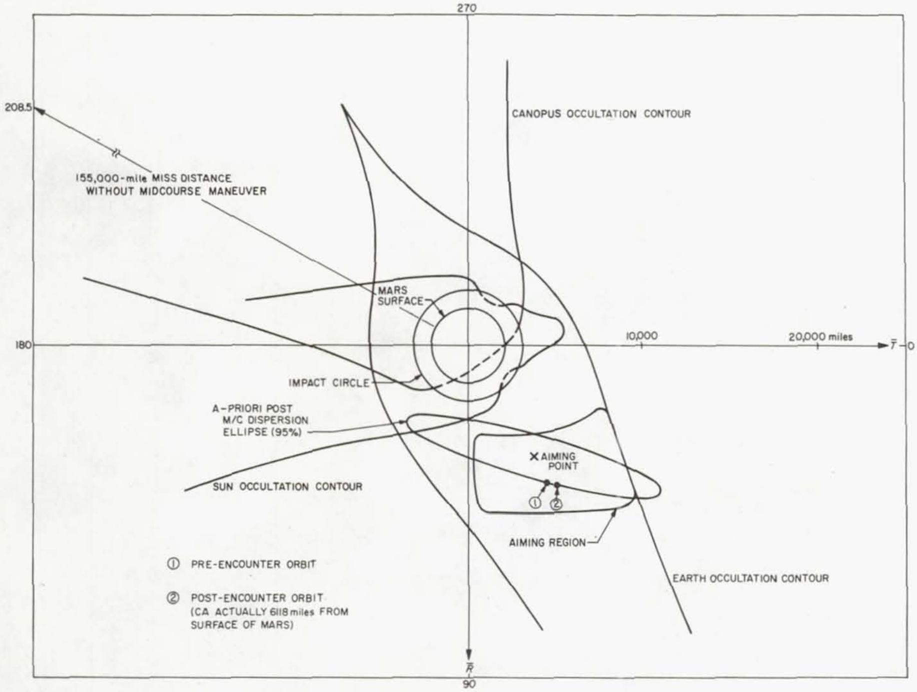


FIGURE 7

Mariner IV Aiming Zone

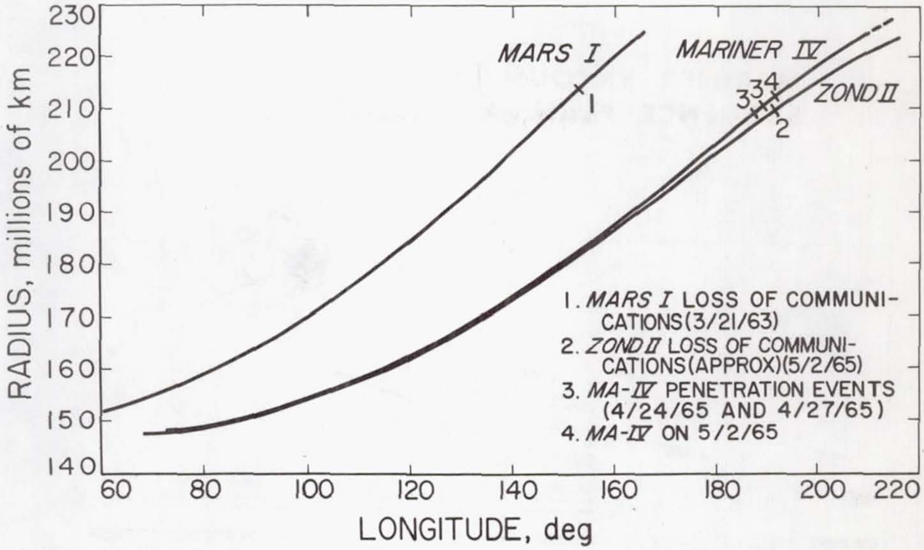


FIGURE 8

Mariner IV, Zond II, Mars I Heliocentric Distance vs Heliocentric Longitude

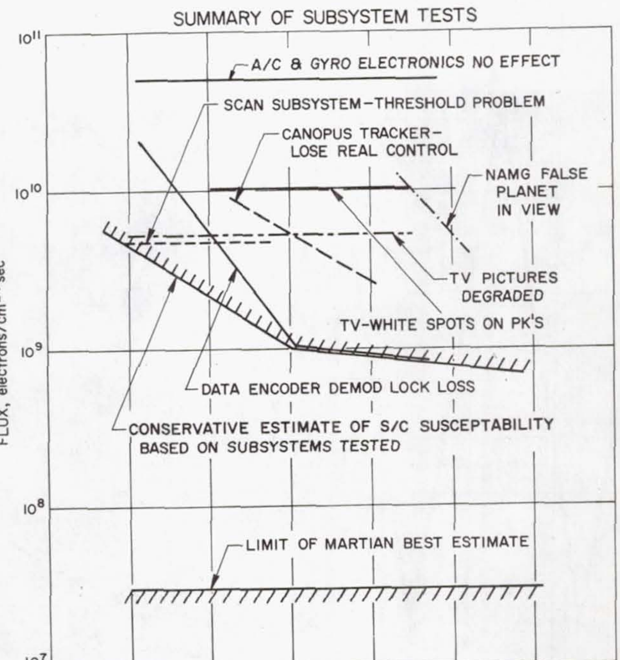


FIGURE 9

Summary Of Subsystem Tests

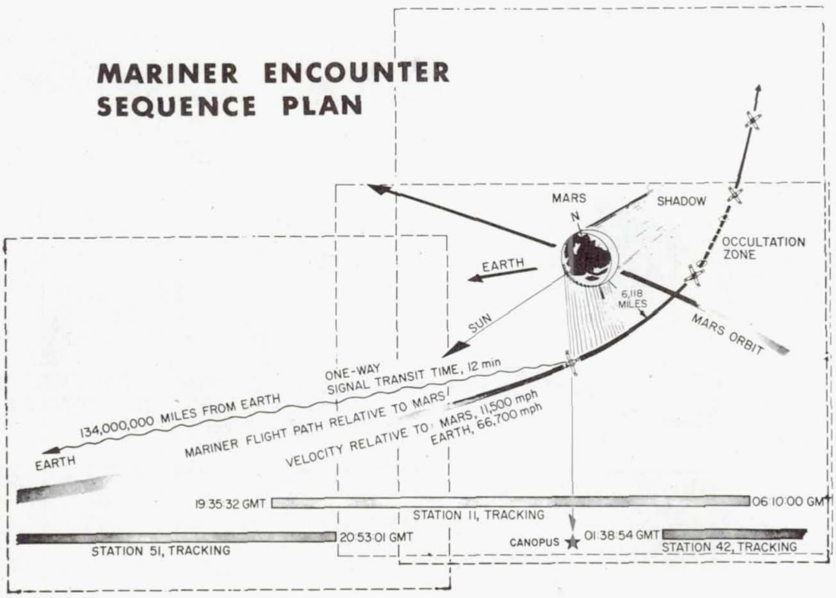


FIGURE 10

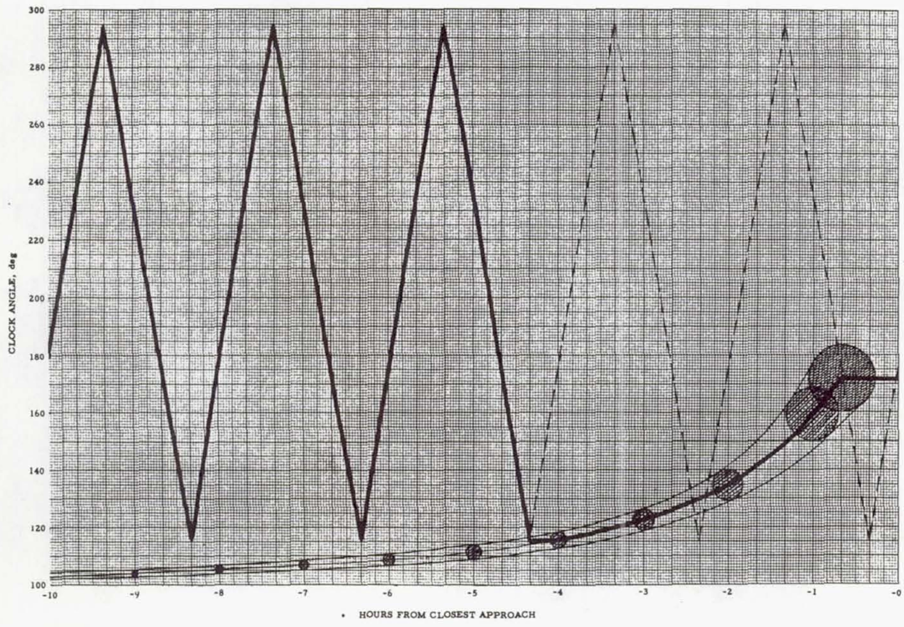


FIGURE 11
Scan Platform Position

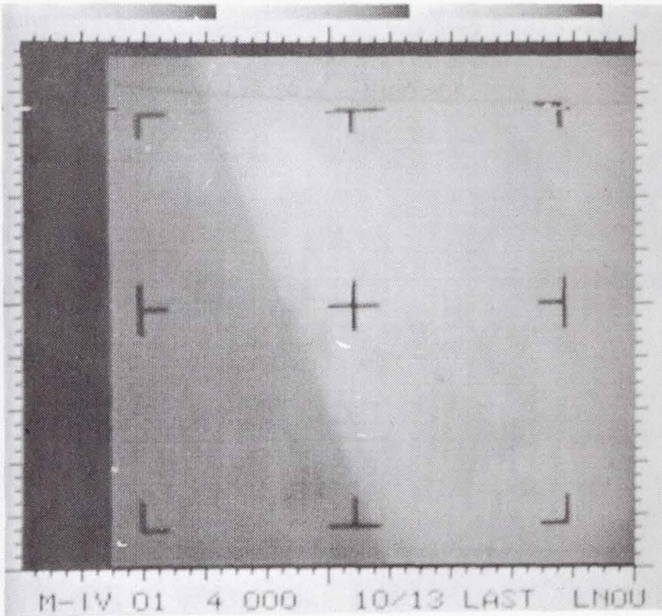


FIGURE 12

First Martian Photo

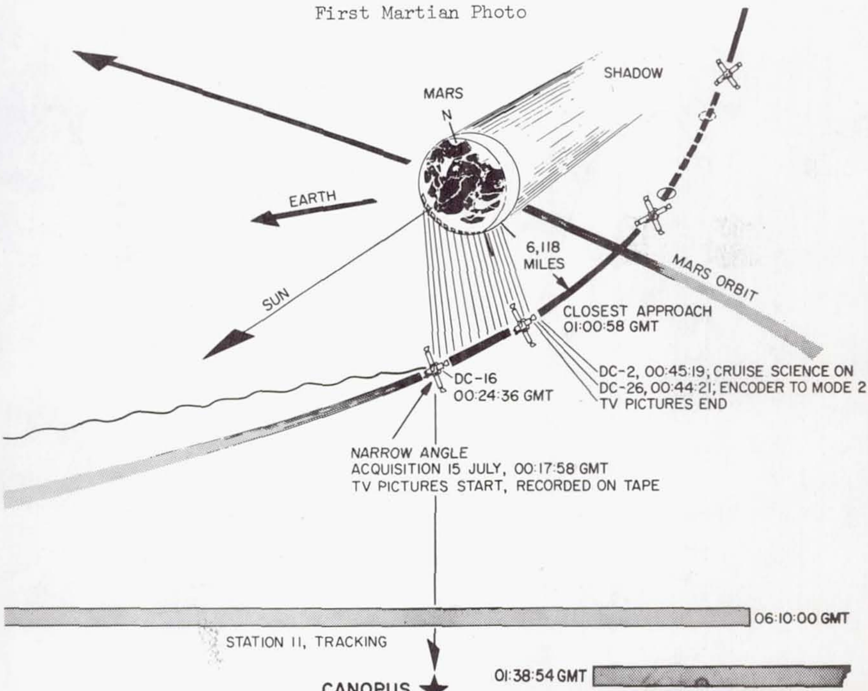


FIGURE 13

Photo Sequence

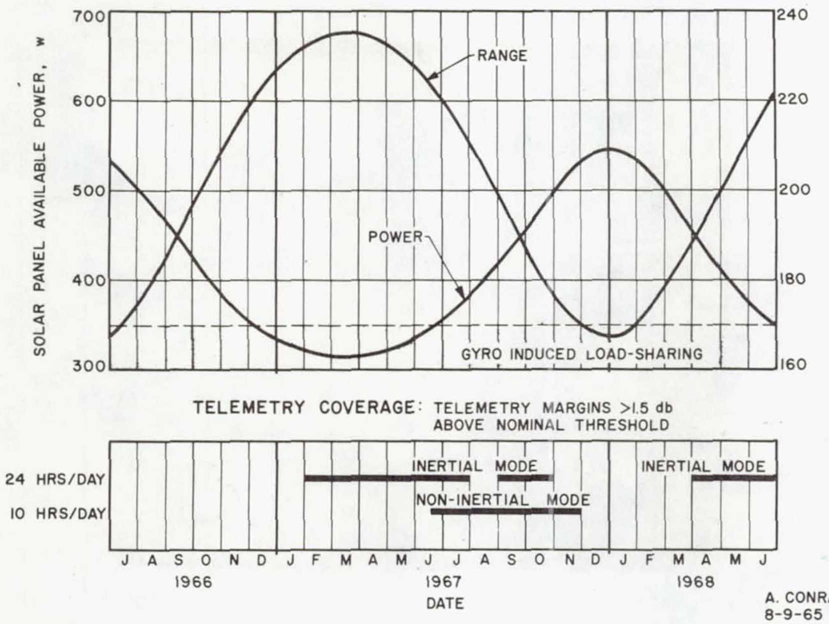


FIGURE 14

1967 Reacquisition

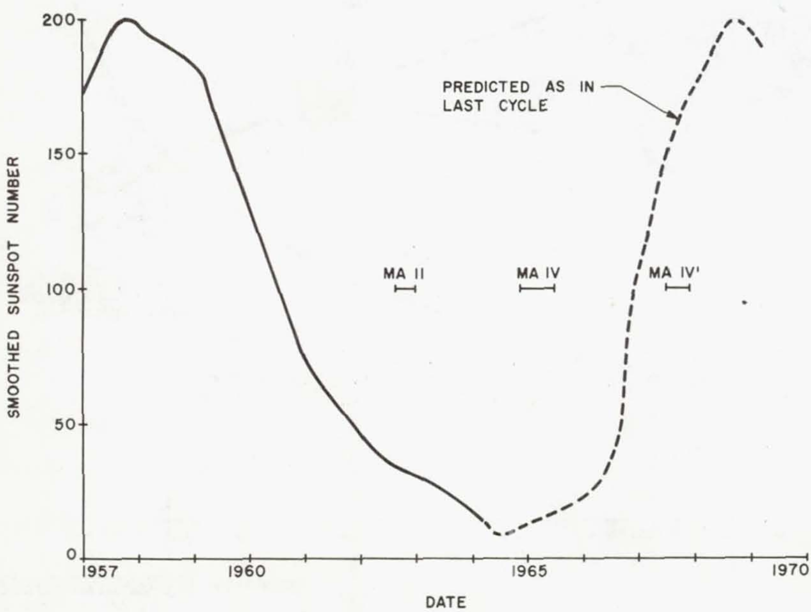


FIGURE 15

Solar Activity and Launch Date

N66 31499

SCIENCE PAYLOADS FROM MARINER II TO MARINER IV

By

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Frank L. Schutz
William G. Fawcett

Jet Propulsion Laboratory

INTRODUCTION

The Mariner program, assigned to the Jet Propulsion Laboratory by the National Aeronautics and Space Administration, consisted of two projects involving four launches designed to investigate the near planets, Venus and Mars. In the first phase, the Mariners I and II spacecraft were launched toward Venus in 1962. In the second, Mariners III and IV were launched to Mars in 1964.

The primary objective of the Mariner 1962 project was to receive communications from the spacecraft while in the vicinity of Venus and to perform a radiometric temperature measurement of the planet. A secondary objective was to make interplanetary field and/or particle measurements on the way to and in the vicinity of Venus.

The primary objective of the Mariner 1964 project was to conduct fly-by observations of the planet Mars during the 1964-1965 opportunity and to transmit the results of these observations back to Earth. Planetary observations were, to the greatest practical extent, to provide maximum information about Mars; specifically, a TV system, fields and particles and cosmic dust experiments were to be carried. In addition, an Earth occultation experiment was to be carried out to obtain data relating to the scale height and pressure in the atmosphere of Mars. Secondary objectives were to obtain experience and knowledge about the engineering performance of an attitude-stabilized flyby spacecraft traveling outward from the Earth and to perform certain field and/or particle measurements in interplanetary space and in the vicinity of Mars.

These two projects, with their somewhat contrasting objectives, have been selected for discussion because of their common generic background and because they were both planetary orientated. Both spacecraft evolved from the earlier Ranger series; they were attitude stabilized and sun-orientated, and used solar cells to supply operating power. This paper is presented to both compare and contrast the two spacecraft, their design, development, testing and operation, from the standpoint of their scientific payloads.

SPACECRAFT

The spacecraft design for both missions was constrained primarily by the capabilities of the launch vehicles. Because of this, significant missions were achieved only through the development of highly integrated spacecrafts. In particular, when the scientific instruments are designed as integral parts of the spacecraft system, optimum configuration weight, power and telemetry are obtainable. This approach therefore permits the greatest scientific instrument payload.

The Mariner Venus 1962 spacecraft was designed for an injected weight of 446 pounds, while the Mariner Mars 1964 spacecraft weighed 575 pounds. The same launch vehicle was used in both instances, but both stages were upgraded and improved for the latter mission.

Mariner Venus 1962

The initiation of the Mariner 1962 project resulted from cancellation of the Mariner "A" project. The Mariner "A" was to be a 1050 pound spacecraft designed to carry approximately 130 pounds of scientific instruments on a fly-by past Venus

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*This paper presents the results of one phase of research carried out at the Jet Propulsion Laboratory, California Institute of Technology, under Contract No. NAS 7-100, sponsored by the National Aeronautics and Space Administration.

in 1962. It was concelled due to lack of launch vehicle capability to inject 1050 pounds to Venus. Therefore, a 50 pound scientific payload was launched on a 446 pound spacecraft (Figure 1). This reduction resulted because the basic spacecraft requires specific equipment to perform the mission regardless of the instrument payload size. These basic spacecraft equipments are termed "mission critical" because they are required to function properly before meaningful scientific information can be obtained. A functional block diagram of the elements of the Mariner 1964 spacecraft is shown in Figure 2.

The change in mission direction necessitated by the smaller spacecraft affected the science payload far more than it affected other subsystems because the science instruments had been configured for a specific mission and were either too heavy or required data handling capabilities not available from the reduced version of the spacecraft. Thus, the entire science payload - the scientific data handling equipment as well as the instruments - had to be redesigned to meet the new launch vehicle constraints and spacecraft configuration.

A crash program was initiated in September 1961 to develop a spacecraft and instruments to meet the revised mission objectives. The Mariner 1962 payload (Figure 3) included reduced versions of many of the instruments originally selected for the Mariner "A" payload. However, in almost every instance the redesign was so extensive that little of the previous developmental effort could be utilized. The numerous anomalies or problems which developed during the spaceflight portion of the mission indicate that this total effort was only partially successful. Many of these difficulties could surely have been eliminated if the design, development and testing schedule had not been so compressed. In spite of this, all mission objectives were accomplished and a wealth of meaningful scientific and engineering data was accumulated.

Mariner Mars 1964

As with the Mariner 1962, the Mariner 1964 Project developed from the cancellation of previously scheduled missions. These were larger spacecraft which had been designed for dual mission capability, either Venus or Mars, during the 1964 launch opportunities. Again, due to launch vehicle availability, it was necessary to reduce the spacecraft weight from approximately 1250 to 575 pounds with corresponding scientific payload reductions.

Also, as in the case of the earlier mission, the scientific payload selected incorporated a number of instruments originally selected for the larger spacecraft. Again, significant modifications of the instruments were required before they could be integrated into the new spacecraft (Figures 4, 5 and 6).

The schedules under which these two projects were conducted are contrasted in Figure 7. There are several significant facets of these schedules that should be noted:

1. The period available for design and development of Mariner 1962 was roughly half that for Mariner 1964.
2. There was no proof test model (PTM) on the Mariner 1962 project; thus, there was no time for major hardware redesign if any serious difficulties occurred during testing.
3. The Mariner 1964 spare flight spacecraft was assembled and tested before shipment to the launch facility, while the Mariner 1962 spare spacecraft was assembled and electrically tested at the launch facility--it had no environmental testing.

SCIENCE SUBSYSTEM

As a result of the Mariner 1962 experience, particularly in those areas concerning scientific instrument integration, specific changes were made in the concept of spacecraft/scientific instrument interface mechanization, use of flight qualified electronic components, instrument operation and control, and testing.

One important requirement for the design of any subsystem is the ability to operate that subsystem without anomaly within the major system of which it is a part. To accomplish this on the Mariner 1964 spacecraft, the concept of an integrated science instrument payload was developed (Figure 8). This concept was based primarily upon the Mariner 1962 experience where excessive problems were encountered with the instruments after they were installed on the spacecraft. To

minimize these problems on Mariner 1964, it was decided that the scientific instruments and their ancillary equipment would be integrated into the spacecraft as a separate subsystem; heretofore each instrument had been a subsystem itself. This, then, allowed the science instruments to be integrated into and tested as a subsystem before assembly into the spacecraft system.

Subsystem Design

The science subsystem design was initiated by establishing the constraints imposed on the instruments by the spacecraft, the constraints imposed on the spacecraft by the instruments, and the characteristics of the instruments. Figure 9 and 10 are typical characteristics of instruments used on deep space probes. Analysis of these constraints and characteristics then led to the design and development of the ancillary equipment required to integrate the instruments into the subsystem and the spacecraft.

For the Mariner 1962 spacecraft, the subsystems supporting the instruments were the data conditioning system and the scientific power switching unit (Figure 11). The data conditioning system performed the data acquisition and conversion functions required to transfer data from the scientific instruments to the spacecraft telemetry system. These functions included analog-to-digital conversion, digital-to-digital conversion, measurement timing and sequencing, periodic calibration and planetary acquisition control. This latter function was performed by examining the output from the microwave radiometer and logically deciding whether the planet was or was not in the radiometer's field of view. The scan velocity and area coverage were then dependent upon the results of this logical decision. The power switching unit, being logically sequenced by the data conditioning system, controlled the power supplied to the instruments.

The in-flight operation of this payload was not as good as had been anticipated. There were a number of anomalies that affected both the measurements and the operations of the subsystem. These involved:

1. The planet acquisition and tracking mechanization.
2. The analog-to-digital conversion technique.
3. The in-flight calibrate sequence.

In spite of these anomalies, the Mariner 1962 scientific instruments, and their ultimate performance, represented a step forward in the design of instruments for deep space probes.

In designing the Mariner 1964 science subsystem, care was taken to preclude repetition of these anomalies. To this end, specific changes were made in the instrument/data system interfaces and, in addition, a planet acquisition and tracking device, the planetary scan system, was established as an independent piece of ancillary equipment. The other ancillary equipments to this science subsystem were the data automation system and narrow angle Mars gate. The primary characteristics of these devices are listed in Figure 12.

The Mariner 1964 data automation system was split functionally into two parts: one operated throughout the flight and controlled all the instruments except the television and planetary scan systems, while the other was energized during actual planetary encounter and controlled the television picture and planetary acquisition sequence. A number of methods were employed to ensure the integrity of signals crossing interfaces between the instruments and the data automation system. No analog signals were brought across any of these interfaces. Instead, analog-to-pulse-width converters were installed in each instrument. When a measurement was to be made, the data automation system sent a read-command to the instrument and started accumulating counts in an internal register. Upon receipt of this command the analog-to-pulse-width converter generated an output signal delayed by a period of time proportional to the analog voltage measured. This signal was transformer-coupled to the data automation system where it was used to inhibit the accumulation of counts in the register. The binary number in this register then represented the digital equivalent of the analog voltage. In addition to the above, transformer-coupling, and separate, tightly-twisted wire pairs were used for the transmission of nearly all pulses between subsystem elements. This system had a number of distinct advantages: (1) No noise could be coupled into analog signal lines to invalidate the measurements. (2) DC ground loops were eliminated, thus reducing interaction among the elements of the subsystem and spacecraft. (3) The effects of electromagnetic coupling in cable harnesses were greatly reduced.

The Mariner 1964 planetary scan system design was predicated upon the requirement to perform a normal sequence of planet searching, acquisition and tracking using internal logic. However, instead of the continuous slow scan across the planet from limb to limb as required during the Mariner 1962 encounter, the Mars spacecraft's scan system was designed to be inhibited as soon as the planet entered the television field of view. This was because the radiometers on Mariner 1962 were used to temperature-map the planet, while the television system on the Mariner 1964 spacecraft required a motionless scan platform during vidicon exposure to prevent picture smearing. An additional feature of the Mariner 1964 design was the incorporation of a ground command back-up as a redundant measure to permit the searching and tracking functions to be inhibited in the event of failures. For operational purposes this mechanization was used during the actual Mars encounter to preposition the scan platform at an optimum angle for the acquisition of meaningful television data.

Testing

If the instrument, subsystem and spacecraft design are to be adequately validated prior to launch, then the testing program must be emphasized to the extent that it assumes equal importance with the design and development program. From Figure 7 it is readily apparent that there was time for this for the Mariner 1964 project, but not for Mariner 1962.

Instrument Testing

The initial bench tests of Mariner 1962 instruments were considerably complicated by the discovery of fabrication errors and the occurrence of early component failures. These problems were virtually eliminated as a result of the rigid electronic component parts screening and quality assurance efforts required for the Mariner 1964 project. After the bench tests and initial calibrations were complete, each instrument for both projects was then required to pass an exacting series of environmental tests prior to spacecraft assembly. One series, Flight Acceptance testing, validated the integrity of individual instruments. A second series, Type Approval, was established to test the adequacy of the electrical and mechanical design. The latter tests, mandatory for Mariner 1964 but only optional for Mariner 1962 because of the severe schedule constraints, were conducted at levels far exceeding expected environmental stresses. In addition, the environmental levels were somewhat different for the two projects, with the Mariner 1964 testing being more rigorous as a result of experience gained with the earlier spacecraft.

The testing of the Mariner 1962 instruments was inadequate in an overall sense because too short an interval was available between instrument design completion and scheduled flight use. This does not mean that they were not qualified instruments, but rather that the extent of the testing performed was not considered sufficient to provide a high level of confidence in their ability to survive the mission.

On Mariner 1964, however, each instrument was thoroughly tested and calibrated. Furthermore, recalibration or calibration verification was performed on each instrument just prior to launch. In many cases, these operations were performed on the spacecraft to establish that the instrument was unaffected by the presence of the spacecraft.

Subsystem Testing

One of the ground rules established for the Mariner 1964 program was that the scientific instruments and ancillary equipment should be tested and integrated into the spacecraft as a single sybsystem. This had been impossible under the schedule constraints imposed by the Mariner 1962 program. In order to test the Mariner 1964 subsystem in the above manner, a dummy spacecraft was fabricated and the instruments and ancillary equipment tested on it before delivery to the spacecraft assembly facility. This accomplished several significant objectives:

1. It verified the interfaces between the instruments and their ancillary equipment.
2. It established intrasubsystem compatibility.
3. It verified the design adequacy of the ground or operational support equipment.

Performing these tests before delivery of the subsystem to the spacecraft eliminated the extensive post-delivery trouble-shooting that had been required on the Mariner 1962 spacecraft.

System Testing

Although the Mariner 1964 testing philosophy attached importance to predelivery tests of the instruments, the primary determinant of performance was the science subsystem's ability to operate reliably after integration into the spacecraft.

The subsystem interface compatibility with the spacecraft was established by performing a series of subsystem-to-subsystem tests. When these tests were completed, the entire spacecraft was given a system test. This involved an elaborate procedure designed to test each subsystem interface and to establish that each subsystem operated without anomaly in conjunction with the other spacecraft subsystems.

While system tests of the previous spacecraft were generally 8 to 12 hours in length, they were extended to almost 60 hours for the Mariner 1964 project. To establish that the science instruments were performing properly, a 30-hour "quiet test" was incorporated into the systems test procedure to allow accumulation of sufficient data for statistical analysis and to standardize the stimuli to the instruments so that long term degradation or drifts could be detected. Since each television picture required 8-1/3 hours to play back from the spacecraft tape recorder, the quiet test period was also used to accumulate data for comparison with that obtained directly from the television and data automation systems during the recording phase.

Each spacecraft was placed in a space simulator for 14 days to establish its functional integrity while operating in a simulated space environment and to verify the design of the temperature control system.

Three of the Mariner 1964 instruments developed problems in preflight testing which resulted from high-voltage breakdown. In each case the failures were detected during thermal vacuum testing of a complete spacecraft.

The first problem involved transients so serious that they would have aborted the mission had they occurred in flight. A breakdown in the high-voltage power supply of this instrument occurred because it had been covered by a polyurethane foam and then conformally coated. The failure resulted because a small pocket of trapped gas accumulated in the area between a high-voltage point and the instrument common each time the instrument was subjected to a hard vacuum. This problem was discovered too late to take corrective action and still have sufficient time to properly re-qualify the instrument for use on the spacecraft; it was deleted from the payload four months before launch.

The second problem was discovered when the plasma probe program sequence skipped steps at random intervals. This occurred whenever the instrument was subjected to a vacuum environment. It was found that air trapped in the silicon rubber spacers between the grids of the plasma probe sensor caused minute high-voltage discharges to occur until the spacers had outgassed. This was resolved by using Kel-F for the spacers instead of the silicon rubber.

The third high-voltage difficulty resulted from a failure to provide venting around the base of a Geiger-Mueller tube. As in the previous case, a silicon rubber compound had been used as an insulator. This time the failure was detected by examining the spacecraft data. Transients set up by arcing of the ionization-chamber Geiger-Mueller tube caused the data from the cosmic ray telescope and the cosmic dust detector to be abnormal. Venting of the insulator eliminated this difficulty.

Other specialized tests conducted on these spacecraft included:

a. A complete spacecraft vibration test in three planes.
b. A measurement of the spacecraft weight and center of gravity.
c. A parameter variation test to ensure that the spacecraft could operate with a plus or minus 10 percent variation in primary voltages and synchronization frequency.
d. A static and dynamic magnetic field mapping test to establish the gross effect of the spacecraft on the magnetometer.
e. A free mode test to observe operation of the spacecraft while using solar power. These results were compared with previous tests to ascertain the effects of the direct access and umbilical cable connections on the spacecraft's operation.
f. Several simulated launches and combined system tests. These were conducted to verify the compatibility of the spacecraft with the Atlas, Agena, Air Force Eastern Test Range, and other agencies supporting the launch.

These tests performed on the Mariner 1964 spacecrafts resulted primarily from experience gained on the earlier Ranger and Mariner projects. The parameter variation and free mode tests, for instance, were not performed at all on the Mariner 1962 spacecraft.

The successful completion of these tests was required before launch and each anomaly was carefully investigated to determine its source and effect on the spacecraft. Just prior to the final system test, a calibration verification of the scientific instruments and a planetary instrument sensor alignment test were performed.

SUMMARY.

The complexity and relative youth of the space program has resulted in certain problems of implementation not normally encountered by electronic designers. Protracted operation of an untended space vehicle is relatively new technology, especially when many of the components and designs used are at or near the state-of-the-art.

The problems encountered during the design and development of the science subsystems for both Mariner projects were similar in their effect on spacecraft evolution. This design evolution was, in turn, predicated upon the manner in which the spacecraft components were assembled. In the Mariner, where the integration of the subsystems into the spacecraft was complicated by launch vehicle and communication constraints, the spacecraft design preceded the science subsystem design. Only in this way could an estimate be made of the total weight and power available for science instrumentation. In future spacecraft, of the Voyager class, these constraints may not be as serious; however, on Mariner they affected the design profoundly.

Another factor that delayed the design of these science subsystems was the type and operational status of the instruments selected and ultimately integrated into the spacecraft. Figure 13 is a comparison of the Mariner 1962 instruments' status at the time they were selected and when they were launched. It is readily apparent from this figure that significant changes were required in the design of the selected instruments. Another influencing factor was the addition of the infrared radiometer to the payload in December 1962, just six months before shipment to the launch facility. This late change forced a modification of the science data conditioning system to accommodate the new data handling requirements. Between the selection of the instruments and final payload integration, significant design and development effort had to be expended to meet the schedule requirements.

Figure 14 is a listing of the Mariner 1964 instruments' status at the time of their selection and at launch. Again, it is obvious that radical changes were required in these instruments before they could be integrated into the spacecraft. Two of these instruments, the television and the magnetometer, had no previous flight experience. An additional instrument, the ultraviolet photometer, developed a serious high voltage breakdown problem and, although the instrument was redesigned to eliminate this deficiency, it was deleted from the payload because inadequate time remained before launch to re-qualify it as a flight instrument.

These comparisons clearly illustrate the flexibility that was required to integrate the science sybsystem into the Mariner spacecrafts. Although flexibility was required primarily because the design and development of the instruments and subsystem lagged that of the spacecraft, the uncertainties in launch vehicle capability were secondary contributing factors. This elasticity was achieved through the design of a science subsystem which integrated the mission constraints, the spacecraft interfaces, and the instrument requirements. The results were instruments and ancillary equipment comprising integrated subsystems capable of providing a maximum of scientifically significant, reducible data.

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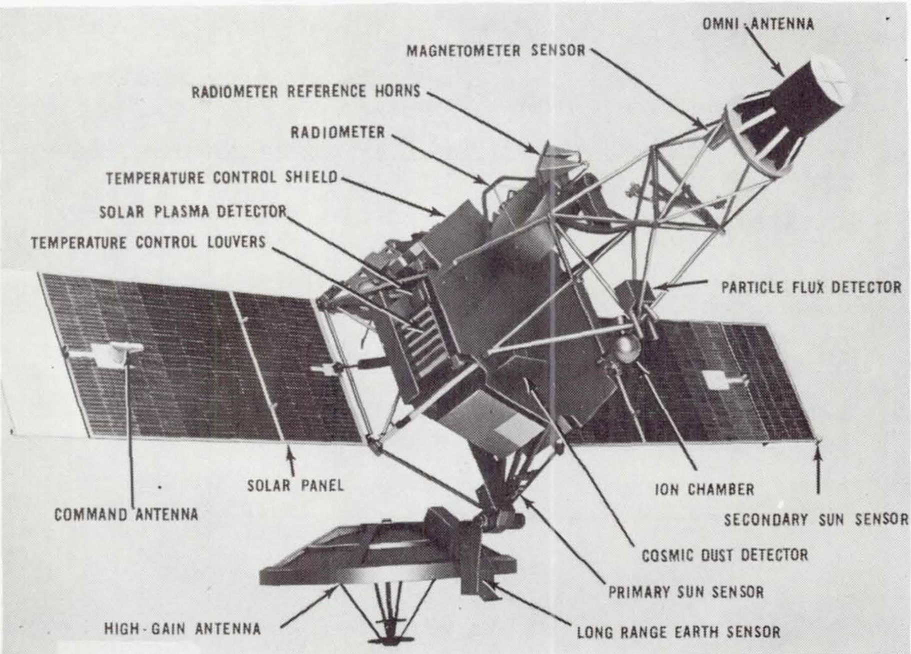

FIGURE 1

Mariner 2 Spacecraft

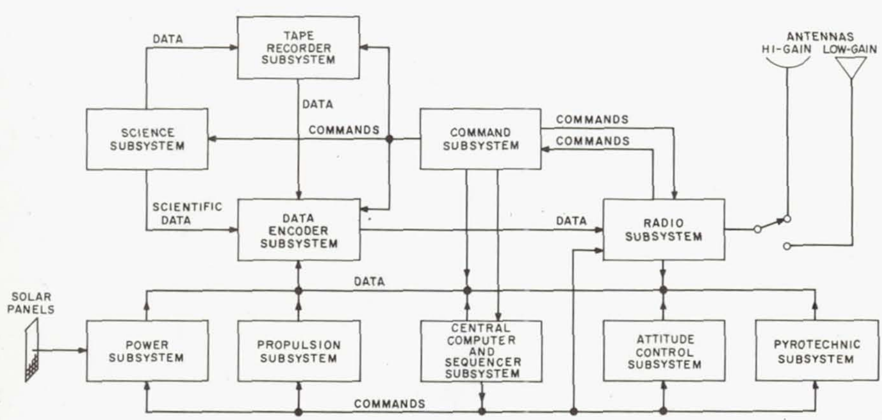

FIGURE 2

Functional Block Diagram of the Elements of the Mariner 1964 Spacecraft

MARINER 1962 INSTRUMENTS CHARACTERISTICS

DESCRIPTION	WEIGHT, #	POWER, W	LOOK ANGLE	PURPOSE	SENSORS/OUTPUTS
Magnetometer	4.70	6.00	Omni-directional	To measure the steady state and varying components of magnetic fields	3 sensors, analog
Ionization Chamber, Particle Flux Detector	2.78	0.40	Omni-directional and 90° cone	To measure charged particle intensity and distribution	4 sensors, digital
Cosmic Dust Detector	1.85	0.08	Into plane of ecliptic	To measure the flux momentum of cosmic dust	1 sensor, digital
Solar Plasma Detector	4.80	1.00	10° cone	To measure the intensity of low energy protons from the Sun	1 sensor, analog
Microwave Radiometer	23.80	3.50	2.0° and 2.5° cone	Temperature mapping of the planet's surface and atmosphere	2 sensors, analog
Infrared Radiometer	2.70	2.00	0.9° square	To determine the fine structure of the cloud layer	2 sensors, analog
TOTALS	40.63	12.98			

FIGURE 3

MARINER 1964 INSTRUMENT CHARACTERISTICS

DESCRIPTION	WEIGHT, #	POWER, W	LOOK ANGLE	PURPOSE	SENSORS/OUTPUTS
Cosmic Ray Telescope	2.58	0.60	60° cone	To measure the flux and energy of alpha particles and protons	3 sensors, digital
Cosmic Dust Detector	2.10	0.20	Plane of ecliptic	To make direct measurements of dust distribution	2 sensors, digital
Trapped Radiation Detector	2.20	0.35	60° cone	To monitor solar cosmic rays and energetic electrons	4 sensors, digital
Ionization Chamber	2.71	0.46	Omni-directional	To measure the omnidirectional flux and specific ionization of charged particles	2 sensors, digital
Plasma Probe	6.41	2.65	30° cone	To measure the distribution, density and time history of the solar plasma	1 sensor, digital
Magnetometer	7.50	7.30	Omni-directional	To measure and investigate steady state and slowly varying magnetic fields	3 sensors, digital
Television	11.30	8.00	1.05° square	To make preliminary topographic reconnaissance of the surface of Mars	1 sensor, digital
TOTALS	34.80	19.56			

FIGURE 4

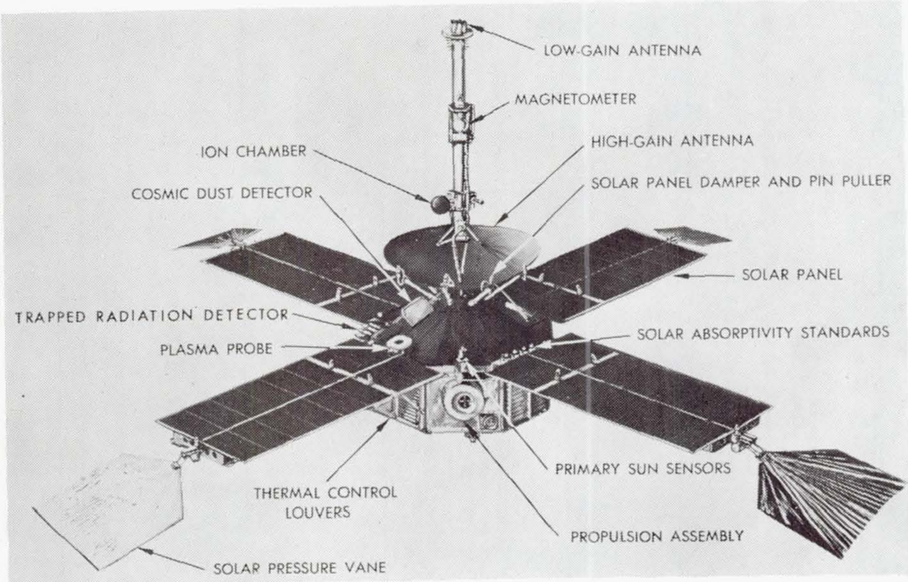

FIGURE 5

Mariner/Mars Spacecraft

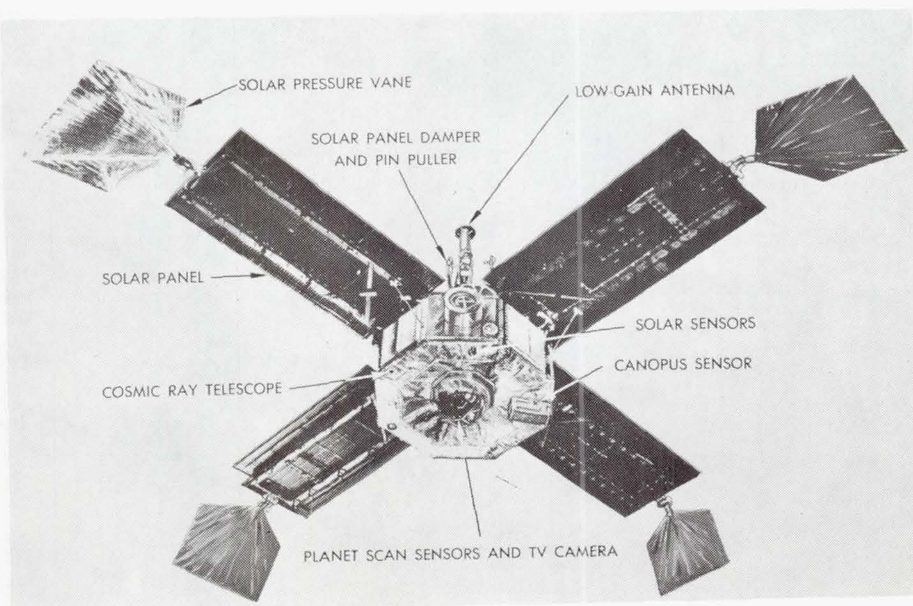

FIGURE 6

Mariner/Mars Spacecraft

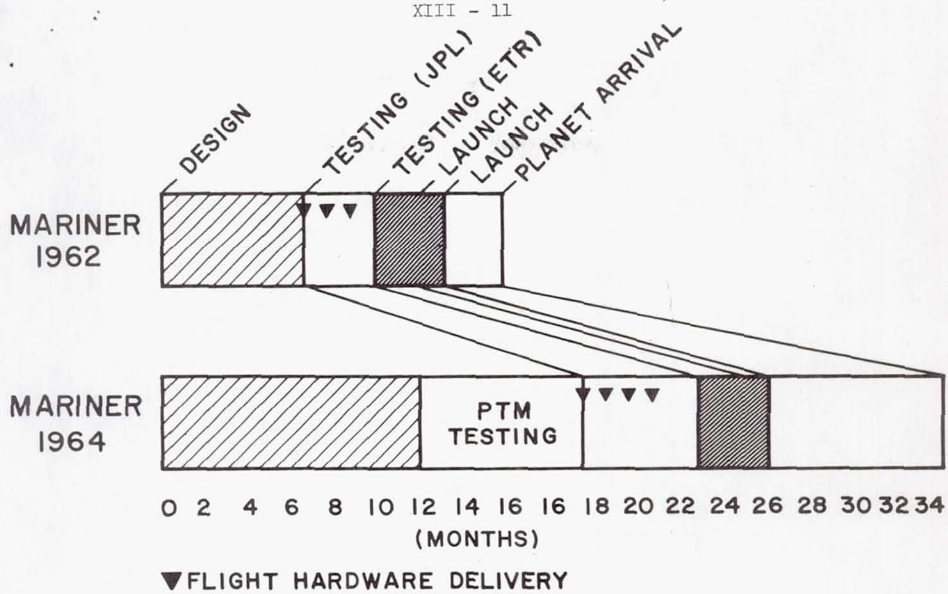

FIGURE 7

Schedule Comparison Mariner 1962 and Mariner 1964

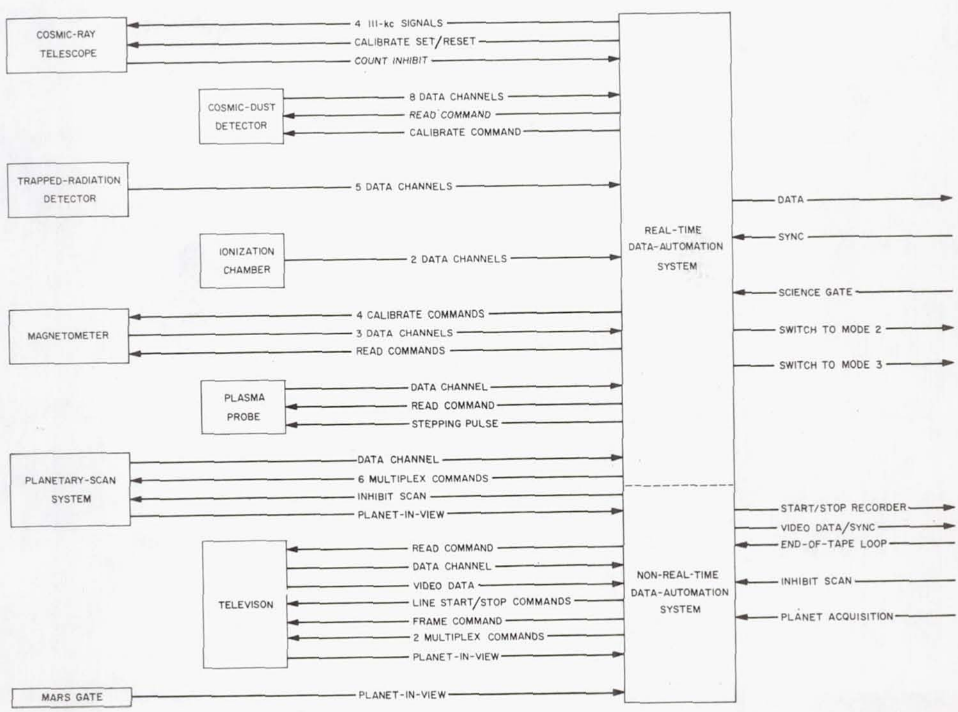

FIGURE 8

Diagram of the Integrated Science Instrument Payload for the Mariner 1964 Spacecraft

GENERAL INTERPLANETARY INSTRUMENT CHARACTERISTICS

1. Constraints on spacecraft materials.
2. Low average data rates.
3. Small internal spacecraft data storage requirements.
4. Broad fields of view.
5. Wide range of operating temperatures:
6. Relatively low weight and power requirements.
7. Special spacecraft mounting requirements.
8. Special data handling requirements.

FIGURE 9

GENERAL PLANETARY INSTRUMENT CHARACTERISTICS

1. Narrow fields of view.
2. High, fixed rate, 2--100K bits per second data outputs.
3. Large internal spacecraft data storage requirements.
4. Precise mounting arrangements.
5. Narrow operating temperature ranges.
6. Special sequencing and control.
7. Relatively high weight and power requirements.
8. Larger effects on spacecraft design.

FIGURE 10

MARINER 1962 SCIENCE ANCILLARIES

DESCRIPTION	WEIGHT, #	POWER, W	PURPOSE	INPUT/OUTPUT
Data Conditioning	6.60	2.20	To provide the sequencing and processing, of scientific data and to control the planetary tracking at encounter	Analog & Digital/ Digital
Power Switching	1.68	----	To control power to scientific instruments.	Primary Power and Digital Control Signals/Primary Power
TOTALS	8.28	2.20		

FIGURE 11

MARINER 1964 ANCILLARY CHARACTERISTICS

DESCRIPTION	WEIGHT, #	POWER, W	LOOK ANGLE	PURPOSE	INPUTS/OUTPUTS
Data Automation System	11.90	6.50	----	To provide the sequencing, processing, buffering and storage of data to realize the optimum scientific value of each instrument.	Figure 8
Planetary Scan System	5.65	7.46	50° cone	To orientate the television camera toward the planet	Digital
Narrow Angle Mars Gate	0.20	(~0)	2.5°x 1.5°	To detect the planet	Digital
TOTALS	17.75	13.96			

FIGURE 12

MARINER 1962 INSTRUMENT STATUS

STATUS AT SELECTION	MICROWAVE RADIOMETER	LAUNCHED CONFIGURATION
4 Analog	Output Channels	2 Analog
28.00 lbs	Weight	23.80 lbs
5.00 W	Power	3.50 W
Breadboard	Status	----
No	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical
	INFRARED RADIOMETER	
---	Output Channels	2 Analog
---	Weight	2.7 lbs
---	Power	2.0 W
None	Status	----
---	Previous Flight Experience	----
---	Configuration Changes	----
	IONIZATION CHAMBER AND PARTICLE FLUX DETECTORS	
6 Digital	Output Channels	4 Digital
6.5 lbs	Weight	2.78 lbs
2.1 W	Power	0.40 W
Prototype	Status	----
Yes	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical
	SOLAR PLASMA DETECTOR	
1 Analog	Output Channels	1 Analog
8.5 lbs	Weight	4.8 lbs
1.8 W	Power	1.0 W
Prototype	Status	----
Yes	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical
	MAGNETOMETER	
3 Analog/3 Digital	Output Channels	3 Analog/1 Digital
6.5 lbs	Weight	4.7 lbs
8.0 W	Power	6.0 W
Prototype	Status	----
Yes	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical
	COSMIC DUST DETECTOR	
2 Digital	Output Channels	1 Digital
8.00 lbs	Weight	1.85 lbs
1.00 W	Power	0.08 W
Prototype	Status	----
Yes	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical

FIGURE 13

MARINER 1964 INSTRUMENT STATUS

STATUS AT SELECTION		LAUNCHED CONFIGURATION
	COSMIC RAY TELESCOPE	
3 Digital	Output Channels	1 Digital
20 lbs	Weight	2.60 lbs
1.8 W	Power	0.60 W
Breadboard	Status	----
Yes	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical
	COSMIC DUST DETECTOR	
2 Digital	Output Channels	1 Digital
8.00 lbs	Weight	2.10 lbs
1.00 W	Power	0.20 W
Breadboard	Status	----
Yes	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical
	TRAPPED RADIATION DETECTOR	
6 Digital	Output Channels	4 Digital
5.00 lbs	Weight	2.20 lbs
1.50 W	Power	0.35 W
Breadboard	Status	----
Yes	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical
	IONIZATION CHAMBER	
3 Digital	Output Channels	2 Digital
4.5 lbs	Weight	2.71 lbs
1.0 W	Power	0.46 W
Breadboard	Status	----
Yes	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical
	PLASMA PROBE	
1 Analog	Output Channels	1 Digital
20.00 lbs	Weight	6.41 lbs
5.00 W	Power	2.65 W
Protytype	Status	----
Yes	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical
	MAGNETOMETER	
6 Analog	Output Channels	3 Digital
10.00 lbs	Weight	7.50 lbs
5.00 W	Power	7.30 W
Breadboard	Status	----
None	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical
	TELEVISION	
2 Analog	Output Channels	1 Digital
22.00 lbs	Weight	11.30 lbs
12.00 W	Power	8.00 W
Breadboard	Status	----
None	Previous Flight Experience	----
---	Configuration Changes	Electrical/Mechanical

FIGURE 14

THE PROBLEMS OF PREPARING FOR NEW SPACECRAFT PROGRAMS

By

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INTRODUCTION

In the hectic months following the enactment of the National Aeronautics and Space Act of 1958, the period during which the organization of NASA was being formulated, the determination of which potential programs could become real candidates for implementation was a very difficult process. The difficulty lay in fulfilling the criteria necessary for implementation, even though the criteria themselves were few in number and readily definable, at least in a gross sense. The primary criteria involved were only two in number and might be listed as follows:

(1) Is there an adequate technological base for the candidate program; and

(2) Can the resources necessary for accomplishing the candidate program be marshalled on a reasonable time scale.

Evaluating the adequacy of the technological base was complicated not only by the lack of direct systems experience but particularly by the lack of knowledge of the operating environment which complicated the generation of hardware specifications. This factor of the unknown space environment was probably the greatest single factor complicating our early programs.

The second criterion, that of marshalling the required resources, was amenable to a more straightforward solution. Our nation's capabilities in management, science, and engineering were directly applicable to the task. The establishment of space exploration as a national goal provided the impetus required.

Looking back at the seven years that have elapsed since those early NASA days, each of us, as individuals, may have a somewhat different opinion on how well we, as a nation, have satisfied these criteria. But the fact does remain that we now have many successful missions under our belts. The region around Earth has been explored extensively by a large number of satellites, to the point where we now are in fact exploiting this region with meteorological satellites such as Tiros and communication satellites such as Syncom. The lunar region has been successfully pierced by the Rangers. Deep space has now been successfully probed by Pioneers and Mariners, at least out to Mars and Venus. Several additional follow-on projects in each of these areas are in some cases not only approved but are close to a launch date.

As a result of this experience, the importance of the two criteria originally used in the selection of candidates for future projects has been largely modified, and several additional criteria are today applicable.

In fact, our technology has now advanced to the point where we can propose more feasible missions than our available resources will permit us to accomplish.

In addition, in contrast to the frenzy of program selection seven years ago, there has now evolved a reasonably orderly progression of events - even though still somewhat frenzied - in the process of program selection.

The purpose of today's lecture is to describe this process of program selection - to indicate the criteria used, to define the efforts required, to list the types of problems involved, and in particular to define the roles of the space scientist and the industrial manager.

SCOPE AND LIMITATIONS

The above introductory remarks have been quite general and almost philosophical in nature in order to provide a suitable background for the discussion

to follow. However, to make the discussion more meaningful and more specific, it is now necessary to bound the area to be covered. Following are the constraints now being applied with some indication of their effect on the generality of the material to be presented:

(1) The project selection process to be described is currently applicable to the area of unmanned missions. The procedures applicable in the manned area are similar in general, but differ in their relative emphasis on individual points.

(2) The process to be described in the remaining sections of this lecture will be emphasized from the point of view of a NASA center, in particular, the Jet Propulsion Laboratory. To this end, a distinction is made between PROJECT and PROGRAM. The term PROJECT is herein used to describe an undertaking which consists of one or more missions of similar or closely related spacecraft launched to a single target. Examples are Ranger, Surveyor, Mariner Venus, Mariner Mars, and Voyager Mars. The term PROGRAM will be reserved for a related series of projects, such as the "unmanned planetary program."

(3) Specific examples illustrating the presentation will be taken from the unmanned planetary program both because of the author's specialization in that field for the past several years and because of its appropriateness to the present conference.

Although the emphasis in the lecture is on the problems of preparing for new spacecraft projects, the presentation will be generated within the framework of a typical functional flow diagram of the project selection process. During the discussion of each functional element, the problem area will be outlined. Then, upon completion of the discussion of a typical project selection process, it will be applied to the major question currently under study by the planners of the unmanned planetary program, viz. WHAT'S NEXT AFTER VOYAGER MARS?

A TYPICAL PROJECT PLAN

Figure 1 illustrates the major phases of a typical unmanned planetary project from the initial idea to the completion of the first mission. Shown also are the time elements associated with several concrete examples.

The phases shown are the evolutionary result of two main factors: the technical activities and the procurement plan. The process shown is applicable particularly to large projects, such as Voyager Mars. For smaller projects it is generally tailored to meet the specific needs. NOTE: Neither the names of the phases nor their exact content has as yet been formalized in any official sense. The definitions below should therefore be treated only as general descriptions of the elements comprising each phase.

With this proviso, the phases may be defined as follows:

The Project Selection phase (Phase A) is the study phase during which an idea is developed to the point of a NASA Preliminary Project Development Plan, a document in which all the data necessary to reaching a go-no-go decision (including technical description of the mission, a management plan, and the resources requirements) is evaluated and a project approval is reached. This phase is not normally a formal project phase.

If Phase A results in a project approval, the beginning of Phase B (Preliminary Design) then marks the formal start of the project. Within NASA the project organization is developed and specific mission designs are initiated for the purpose of developing a project definition. If a prime contractor is to be selected, as is now the normal case, Phase B then serves as the competitive period during which two or more contractors are funded to do a design study of the mission and to develop plans for the hardware phase. Implementation of Phase B does not necessarily imply that the project will continue. Upon the completion of Phase B, a NASA review determines whether the results justify continuation of the project.

If the approval milestone of Phase B is successfully passed, then Phase C (System Design) is initiated. Phase C has two primary objectives: (1) to assimilate the technical results of the various studies of Phase B into a set of functional specifications which is to serve as the Project Definition and (2) to complete the competitive aspects of the procurement and prepare the successful contractor for the operations phase. A NASA review is accomplished

before initiation of Phase D; however, Phase C is not generally begun unless there is reasonable assurance of Phase D follow-on.

The Operations Phase (Phase D) consists of the design, fabrication, test, launch, and mission data processing.

The following notes are pertinent to the time scale examples shown on Figure 1.

The Mariner Venus mission, whose launch occurred in August 1962, was truly a crash program. The Preliminary Design, System Design, and Operations phases were all combined into one effort lasting nine months. The duration of the Project Selection phase is more difficult to calculate precisely: the actual pertinent system study was accomplished in a few weeks. However, system and subsystem characteristics, and in some cases, actual hardware were taken from the Ranger and earlier Mariner projects which had been under way for some two years. The effective duration of the Project Selection phase was thus somewhat between two and 24 months.

The Mariner Mars project definition resulted primarily from the Mariner Venus concept, but modified as a result of the change in target. Several earlier studies of Mars spacecraft provided additional inputs.

The project phases as defined above are being applied directly to the Voyager project. First studies began in late 1961 and continued until project go ahead in late 1964. The competitive portion of the Preliminary Design phase of the Voyager bus has just been completed and the Project Review is under way. The lander portion of the Voyager is currently completing its Project Selection phase and will shortly enter the Preliminary Design phase.

THE PROJECT SELECTION PHASE

With the above introduction, the remainder of the lecture will be devoted to a discussion of the Project Selection phase.

Figure 2 is a functional flow diagram indicating the major elements of the Project Selection phase of a typical space project.

In general, this phase consists of a series of iterated design studies beginning with a sample set of broad scientific objectives and payloads, which are coupled with the pertinent constraints to serve as inputs to a parametric analysis. The parametric analysis attempts to define in a gross sense the implications of the objectives and constraints on the spacecraft and related systems. The resulting criteria are then used to select "preferred" missions and to refine their objectives for use in specific conceptual design studies. These design studies need to be conducted in sufficient depth to determine the feasibility of the approach. With sufficient iterations, a "preferred" concept emerges, which can then be used as a basis for a gross project plan. Several project plans are then evaluated and compared, and upon a project recommendation a Project Proposal is generated and used in the determination of whether project initiation is justified.

The next five sections below describe in more detail the pertinent elements of each of these steps in Project Selection process.

MISSION SELECTION

For the past four years, the majority of the planning effort expended in the Advanced Planetary Program has been devoted to the Voyager project. Within the Voyager project, the major concentration has been on the planet Mars. The primary reason for this concentration has been the obvious one: Mars has been the most likely candidate in the solar system to have supported some form of life.

Today, the Voyager Mars project is well on its way toward hardware approval. The attention of our planning staffs is consequently shifting from Mars to the other planets and from the Voyager concept alone to other concepts as well as Voyager. The critical question being examined is, of course, "What should the next project be after Voyager Mars?"

The answer is not an obvious one. Figure 3 illustrates the major criteria being applied to the determination of the most likely candidates for the next project.

The gains in knowledge of our solar system which can be obtained by means of developing technology of this Space Age are potentially so vast, that at least to the first order of approximation, it seems almost impossible to obtain any substantial agreement on a priority listing of even the major targets on

the basis of scientific desires alone. But even if it were possible, the addition of the other criteria pertinent to the making of a go-no-go decision would likely result in a sequence of missions that would be different from the scientific priorities.

To permit the project selection process to proceed in an orderly fashion with proper respect for all pertinent criteria requires the following information from the scientific community <u>for each potential target</u>:

(1) A list of scientific objectives, preferably in some order of priority;
(2) A list of experiments capable of meeting each of the objectives;
(3) A list of instruments which can be used to mechanize each experiment;
and
(4) A series of integrated payload packages, varying in complexity from the simplest but still useful to the most complex but still feasible.

The determination of the mission mode (flyby, orbiter, lander, or some combination) required to implement each integrated payload package and whether it is simple or complex is necessary at the outset, and, in general, is a simple evaluation. An interesting exception to the general simplicity of this determination currently exists in the Voyager Mars project. The case in point concerns the objective of mapping the planet in the visual spectrum. During the Phase A Voyager studies, it had been determined through a sequence of tradeoffs, that the optimum approach to meeting this objective was to place into an elliptical orbit about Mars a television-equipped spacecraft, to program the spacecraft to take a handful of pictures during a brief period near its periapsis point and to use the remainder of the orbit to transmit the pictures back to Earth. For a period of several months in orbit, such a technique seemed capable of mapping a significant portion of the planet's surface.*

More recently, it has been recognized that the propulsion system necessary to establish an orbit about Mars, may at some opportunities be more than sufficient, with a suitable choice of flyby trajectory and a gravity-assist from Mars, to return the spacecraft to the near vicinity of Earth. If this is so, it should be theoretically possible to communicate to Earth a larger amount of scientific information as a result of the vastly reduced communication range.

The practical mechanization of such a mission, and its advantages and disadvantages over the orbiter mode are still to be examined as one of several alternate modes of the Voyager project.

When an estimate has been made of the mission modes and the complexity required to implement each integrated payload package, it then becomes possible to estimate, generally on the basis of previous related design studies, gross system and subsystem requirements for each spacecraft. By comparison against available technology, an estimate can then be made of the gross feasibility of each mission. Such an evaluation typically reduces the total number of mission candidates acceptable for an early project start. A thorough and accurate evaluation of available technology is therefore important.

A further reduction in the candidates is accomplished by comparing estimates of funding, manpower, and facilities required against available resources. It was primarily this element of the project selection process which eliminated serious consideration of the implementation of the large Automated Biological Laboratory on Mars in the 1969 and 1971 time period in spite of its scientific justification and the concerted effort on its behalf.

As a result of the above processes, an ordered list of potential candidates for project start becomes available. This list is then used in the generation of a Long Range Plan. In the unmanned planetary program, and in fact in all of the programs under the cognizance of the Office of Space Science and Applications, the list of candidates is continuously under review to reflect changing conditions, and the Long Range Plan is printed annually as the OSSA Prospectus for guidance to NASA planning personnel.

The Long Range Plan in turn is then used to determine the sequence in which mission studies are to be conducted. Those candidates which are listed

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*Refer to lecture on Orbital Spacecraft Design Proglems by Dr. P. K. Eckman.

for an early start but whose feasibility has not been thoroughly evaluated get first call for more detailed technical study. The type of study to be conducted is determined by the degree of knowledge already existing. For example, the requirements of a second-generation Venus mission, either a flyby or an orbiter, are felt to be understood, at least in a gross sense, as a result of previous studies. The next study to be accomplished in this area is therefore planned to be a Conceptual Design. In contract, a first-generation Jupiter probe has not been studied in depth, and therefore is planned to enter a Parametric Analysis stage.

PARAMETIC ANALYSIS

The primary objectives of a parametric analysis of candidate missions is the attainment of a cursory understanding of mission characteristics over a broad range of parameters in order to uncover potential problem areas for later, more detailed studies. The parameter normally varied is the scientific objective.

A current example of this type of study is the Jupiter flyby study which is being initiated by JPL through an industrial contract. Pertinent quotations from the Statement of Work are as follows:

"(a) The Contractor shall perform a feasibility study to develop spacecraft design concepts for a "flyby" mission of the planet Jupiter. The study shall consider a range of alternate design concepts for accomplishing the successive mission objectives listed below within the applicable design constraints:
- (1) Interplanetary and planetary measurements of the spatial distribution of particles and fields...
- (2) Measurements of the planetary atmosphere of Jupiter...
- (3) Measurements of the physical properties of Jupiter...

(b) In the performance of this study the Contractor shall:
- (1) Develop the conceptual designs for spacecraft systems for each of the objectives above by accomplishing the following:
  - (i) Establishing the functional requirements for spacecraft systems to perform the mission.
  - (ii) Forecasting the applicable state-of-the-art for the time period considered.
  - (iii) Synthesizing the appropriate system concepts.
  - (iv) Identifying the problem areas and indicating approaches to their solution.
  - (v) Reviewing the system concepts in terms of the Mariner Mars '64 spacecraft system design.
- (2) Provide a description for each of the systems developed above...
- (3) Provide estimates of schedule, cost, and probability of success, including success of partial missions, for each of the systems above, and indicate the trade-offs involved..."

Since it is generally only the broad results that are of value in such a study, the depth of analysis is controlled by limiting either the funding of the study to approximately $100,000 or the manpower to approximately 25 man months.

In general, studies such as the above are contracted out to industry rather than done in-house at NASA. The capability to accomplish a parametric analysis of this type exists in many companies and satisfactory results are usually obtained.

Inputs Required for a Parametric Analysis (Figure 4)

A target model is required for two reasons: to determine the dynamic range required of the scientific instrumentation and to define the environment in which the spacecraft must operate. In our studies of Mars landers during the past two years, it was quite a feat of juggling to keep lander design parameters in step with the changing estimates of the atmospheric densities. Now that the atmospheric model appears to be stabilizing, our main concern lies in determining a reasonable meteorological model (in particular, surface winds) and a terrain model, both of which are critical to the landing dynamics and post-landing operations.

Sample scientific payloads generated earlier for the selection of missions are now used as inputs to the parametric analysis.

Certain trajectory and orbit determination parameters, such as the aiming point at the target, the expected dispersion about the aiming point, the direction of approach of the spacecraft and hence the target look angles, and whether the spaçecraft approaches the target on the sunlit side or the dark side, have a very early bearing on the tradeoff decisions of a mission study. As a consequence, and since it is possible to do so shortly after the basic mission mode has been determined, typical trajectories are run before the engineering team begins its deliberation, and the parameters are used as inputs to the study.

Forecasting the applicable state-of-the-art for the projected time period of a potential mission is one of the more difficult and more argumentative study parameters. It is made particularly difficult in parametric evaluation studies, which generally encompass a series of missions, implying a long interval from the first launch to the last. It should be possible, therefore, to allow for conversion to more up-to-date equipment as a project advances. However, until the problems of spaceflight are far better understood than they are now (and my personal opinion is that several more years are required to reach this point), the main guideline that should be, and is, applied is the old adage: "IF IT WORKS, DON'T FIX IT"

During the development phases of both the Mariner Venus and the Mariner Mars mission and after the successful conclusion of the Mariner Venus shot, a hard and fast rule that was used was that no change in design was permitted unless thoroughly justified - and the process of justification required that the old component either must not be applicable to the new mission or must show strong indication of potential failure, rather than just the fact that the new component probably would operate more effectively or would increase the general capability of the spacecraft.*

What is an even more applicable lesson to our planning activities for future projects is the fact that the missions flown to date in lunar and planetary projects have not yet demonstrated that the above approach is ultra conservative. It apparently is still true that the difference between success and failure appears to be a very thin line.

How to predict the state-of-the-art for a given time period in the case of specific hardware items is a process that is well understood, although agreement with any single individual's subjective estimates is not necessarily universal. The most totally-applicable source of hardware state-of-the-art information is of course the on-going space programs of both the National Aeronautics and Space Administration and the Department of Defense. Although I know of no central office in either agency which can supply a master list with sufficient detail to be useful for mission studies, the quantity of ongoing space programs is relatively small in number, and it is fortunately possible to obtain pertinent information for a given subsystem by contacting a sub-infinite number of offices. In the case of JPL mission study contracts, detailed design data on appropriate, successfully-accomplished space missions, such as the Mariner, is supplied to the successful bidder as reference data.

A second source of hardware state-of-the-art information is the research and development programs. Substantial efforts in this area are sponsored not only by NASA and DOD but also by the Independent Research and Development programs of a large number of universities and industrial concerns. These are fruitful sources of information but they suffer from the following limitations:

(1) because of the large number of cognizant offices, it is difficult to reach completeness and thoroughness;

(2) almost by definition, individual tasks are generally stopped somewhat short of flight qualification and therefore require some subjective estimates of their total applicability;

(3) in some cases, particularly in university- and company-sponsored efforts, the special constraints affecting survivability in the space environment are not properly applied either because of a lack of recognition of their impact on the design or on a policy decision of taking as many short cuts as possible to reach the stage at which NASA or DOD might be convinced to assume sponsorship of further development;

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*See a more comprehensive discussion of this point in the lecture on "Review of Mariner 1962 and 1964 Spacecraft Design and Operational Planning" by Dan Schneiderman.

(4) not all of the requirements of future projects have as yet been defined, and therefore there are areas in almost all technological disciplines for which even research has not yet begun. Even in a single program such as the unmanned planetary program, past mission studies have barely scratched the surface of possibilities. However, this lack has at least been recognized and action has been taken to correct it. Specifically, the Office of Advanced Research and Technology at NASA has recently organized a Mission Analysis Division, reporting to NASA Headquarters but physically located at Moffett Field, California, near the Ames Research Center, for the specific purpose of examining distant-future missions to determine requirements of new technologies which can then be implemented into NASA's Advanced Research and Technology program.

The preceding discussion concentrated on a review of the principles of defining state-of-the-art as applied to a subsystem hardware. A simplified summary of the discussion might be that the principles are by-and-large fairly well understood but their application to any future situation involves subjective judgments and therefore may result either in an overly-conservative approach or, as is more frequently the case, in an underestimate of the risk involved.

There is an aspect of technological state-of-the-art other than subsystem hardware which suffers not only from the same limitations, but additionally from an all-too-common lack of recognition that it is even related to technological state-of-the-art. This is the area of SYSTEM ENGINEERING.

Probably the most prevalent single shortcoming of the many mission studies that we have reviewed the past few years, is the treatment of system engineering as a management concept rather than as the technological discipline that it is. Making tradeoffs between subsystems is often considered merely a technique for performing a system design. Conducting a system design by first limiting subsystem hardware capabilities by the constraints of state-of-the-art and then trying to maximize the net payload by this limited definition of system engineering can, and frequently does, result in a spacecraft system whose hardware elements are gratifyingly simple, but whose <u>operational sequence</u> is a nightmare compared to systems with which we have had actual flight experience.

Let me cite a few specific examples to clarify what is meant by technological state-of-the-art in systems engineering:

(1) Our nation has chosen the path of active, continually powered spacecraft for planetary missions. The Russians have apparently chosen the path of dormant spacecraft, activated only upon interrogation. Certainly these are two different approaches to system engineering. Equally certainly, our state-of-the-art in active spacecraft is far advanced over the dormant. If, therefore, one of our advanced missions wanted to take advantage of the potential savings in spacecraft power sources or to limit the allocation of DSIF networks to be a single station, these advantages would need to be balanced against the additional risks associated with our lack of knowledge of and experience with the dormant concept.

(2) I have little doubt that there will come a day when unmanned planetary spacecraft can be designed with the same payload-to-bus relationship that exists today in our buses, trains, and commercial airlines, viz. that a standard spacecraft bus will be designed which will be capable of accepting a significant number of different scientific instruments, mounted to the spacecraft at Cape Kennedy shortly before takeoff. I have little doubt there will come such a day. But our experience to date indicates that such a day is many years away. For the university researcher who still balks at the requirement of making his experiment selection such a long time before launch, let me refer you to the detailed and specific discussions of today's state-of-the-art in system integration in the lectures of Mr. Schneiderman, Dr. Meghreblian, and Dr. Eckman.

(3) Another aspect of system engineering which has associated with it an applicable state-of-the-art is the tradeoff of on-board programming vs. Earth-based commands. In those situations in which both methods are at least analytically feasible, two separate questions need to be answered, viz. (1) if the operation to be mechanized is a link in the chain of operations under nominal conditions, how does one decide whether to use on-board programming or an Earth-based command or to

mechanize both possibilities, with one serving as a backup to the other, and (2) what criteria should be used to determine to what extent either on-board programming or Earth-based commands should be incorporated into the system to provide for nonstandard or failure-mode operation. In either of these questions, only partial answers can be obtained analytically. Total answers are a complex function of subsystem characteristics and reliability, the system definition and mechanization, and engineering judgment - which is probably just an admission that the on-board computation vs. Earth-based commands question is really not very well understood. For a description of today's state-of-the-art in this technology, let me refer you to pertinent portions of Mr. Schneiderman's lecture on the Mariners.

The only comment which needs to be made about applying the constraints of launch vehicle characteristics is a plea directed to the industrial managers who are or will be responsible for parametric studies of the type under discussion. The plea is as follows: Do not apply launch vehicle constraints to such parametric studies until very late in the study.

The natural tendency in any mission study, parametric evaluation or design concept, is to select the required launch vehicles as soon as possible so that configurational layouts can proceed with defined dimensional constraints. The argument against a premature decision is the real possibility that later decisions which should be purely mission-oriented tend to become affected by the launch vehicle decision.

Let me cite as an example a trap that we outselves fell into. Late in 1961 we started our first mission study of the Voyager concept, under the very tight time scale of an expected first launch in 1965 or 1966. Our general concept of the Voyager was that of a combination orbiter/lander. However, we had no data on which to base an estimate of the weight required for such a mission.

To generate such an estimate under the tight time scale and certain manpower restrictions, we decided to concentrate on a single-point design which would provide the data by extrapolation, if necessary. The single point we selected was what we called an "ideal" orbiter - an orbiter capable of containing all the scientific instruments which could be used on an orbiter. The result of our studies was a 5,000 pd net spacecraft, with an additional 25,000 pds of propulsion to produce a circular orbit.

The only launch vehicle capable of sending 30,000 pds to Mars was the Saturn V, which even had an excess capability of 25,000 pds, which could then be available for landers. It didn't take much of an argument to convince ourselves that a Saturn V represented at best an ultimate Voyager, and not a first generation. From our "ideal" orbiter exercise, we were reasonably assured that a pullback to 2,000 pds in orbit with an additional 2,000 pds of propulsion to produce an elliptical orbit still represented a significant mission. We estimated that an additional 4,000 or 5,000 pds would provide sufficient capability for a lander program. Only one launch vehicle was on the books to provide such capability, and on that basis the Saturn I-B with an additional upper stage similar to the Centaur was considered adequate for the mission.

In preparing our justification for such a launch vehicle, we looked briefly at using two smaller vehicles, such as the Titan III-C, for separate launches of the orbiter and lander but quickly convinced ourselves that although the payload capability of two Titans very nearly equalled a single Saturn I-B/Centaur, the additional complexities of separate launches, the weight penalties of two separate buses, and the tracking of two separate spacecraft for several months were not outweighed by any advantages. For these and other considerations the Saturn I-B/Centaur was selected for Voyager.

The Voyager schedule was delayed shortly after this analysis, and we had time to go back and review our decisions in depth. We discovered that in our haste to settle on a launch vehicle, we had underestimated the penalty paid by the orbiter structure to permit it to carry a lander. It now appeared that optimizing separate orbiters and landers for Titans provided appreciably more scientific payload than a combined mission on a Saturn I-B Centaur. This greater payload could now be balanced against the disadvantages and would at least partially outweigh them.

The moral of this story is that although we had recognized that the advantages of a combined orbiter/lander were purely mission-oriented, our application of this factor to the Voyager problem led to conclusions which, in retrospect, carry much less weight in determining the desired system characteristics.

To complete the story, there are other arguments, such as larger ultimate capability for a pure lander, etc., which still give the Saturn I-B/Centaur an edge over two Titan III'C's. So fortunately our oversight was not a major factor in the decision.

Outputs Desired from a Parametric Analysis

A parametric analysis of a series of missions, such as the Jupiter-flyby-with-various-objectives example quoted earlier in this section, should be accomplished in sufficient depth over the entire range of parameters to permit the following decisions to be made and actions to be taken (Figure 4).

Enough supportable data should be provided for each point on the curve in the areas of capability, complexity, schedule, cost, and probability of success to permit selection of a "preferred" mission or sequence of missions for further study. Design concepts or configurational layouts are not of primary importance per se. Their value lies primarily in their use as a mechanism for indicating the depth accomplished in the study and as a framework for uncovering tradeoff points which might otherwise escape notice.

The study should indicate the ranges of applicability of existing launch vehicles, and as a corollary, the areas in which new launch vehicles are required.

The study should define clearly the assumptions made in forecasting the state-of-the-art, including that of system engineering. This should lead into a discussion of potential problem areas, indicating possible approaches to their solution, so that where necessary, appropriate research and development can be initiated.

CONCEPTUAL DESIGNS

If a parametric analysis of a series of missions results in one or more missions with overall properties that appear desirable for project implementations, these selected missions then are submitted through a conceptual design process. The primary objective of the conceptual design process is the evaluation of the technical characteristics of the selected mission(s) in sufficient depth to permit preparation of a gross project plan. The conceptual design is the final technical study before implementation of a project. It should be conducted in sufficient detail to either confirm or definitively revise the properties assigned to the missions during the parametric evaluation process. It should also be conducted in sufficient detail to assure feasibility of at least one proposed approach.

The elements of the conceptual design process (Figure 5) are almost identical to those of the parametric evaluation process. However, differences in emphasis exist within practically all elements.

Inputs Required for Conceptual Design

When the "preferred" missions are selected from the parametric evaluation, the accompanying data determines the selection of launch vehicle. The NASA center cognizant over the selected vehicle can then supply definitive data. This includes payload capability curves, launch injection accuracies, dynamic envelope restrictions, shroud characteristics, permissible range of spacecraft center of gravity location, description of launch environment, description of launch pad operations, and any other characteristics which might affect spacecraft design, such as mechanical and electrical interfaces. These factors define a set of boundary constraints on the spacecraft design and are hence used as inputs to the design studies.

The target model generated earlier is also applicable here, although it may have to be expanded in some particular area as a result of special sensitivity to some spacecraft requirement uncovered by the parametric evaluation.

When the mission has been evaluated parametrically, the functional requirements of the spacecraft defined, and the "preferred" missions selected, it may become evident, for optimization of such characteristics as midcourse correction capability, choice of guidance techniques, or target approach geometry, that a new set of trajectories, with somewhat different characteristics, is required. Otherwise, the same criteria apply as for the parametric evaluation studies.

As indicated previously, one of the main objectives of a parametric analysis is to determine the requirements placed on a mission by each of several scientific objectives. The selection of "preferred" missions upon completion of the study then involves, among other things, the selection of one or more "preferred mixes" of individual scientific objectives. These packages of objectives are then converted through their related experiments to integrated payloads of instruments. It is these modified payloads which are now used as inputs to the conceptual designs.

The same criteria apply to technological state-of-the-art here as in the parametric analysis.

Design Studies

A conceptual design study is conducted in much the same manner as a parametric evaluation. However, now the number of alternative concepts is reduced, and the inputs, as shown above, are better defined. As a result, for the same expenditure of resources, it is possible to obtain the greater depth of analysis that is required in the conceptual design phase.

Additionally, this is the kind of study with which the aerospace industry is most familiar and we find it generally fruitful to subcontract the effort.

In recent months we have become somewhat enamored with the value of a somewhat different kind of system study, which might be called "an evaluation of fundamentals." It is a study whose timing is perhaps better suited to the parametric evaluation phase, although its scope is even more restricted than that of a conceptual design study. It might be best described as the study of a mission whose objective is the mechanization of some very fundamental goal by a concept which requires a minimum number of sequential operations for complete success.

An alternative definition would be to say that it is the study of a mission whose objective is "to get there and prove that it did, in the simplest and most reliable manner."

This technique was first applied at JPL to a Mars capsule study a year ago in order to evaluate the magnitude of the effect of the reduction of surface pressure from 85 mb to 10 mb. The fundamental objective of the mission studied was to "land and communicate" - no science, just land and communicate.* During the course of the study, as each decision point was reached, the primary criterion applied was to make the decision which resulted in the fewest number of operational steps for complete success.

Although the primary intent of the study was to gain an understanding of the fundamental processes involved in landing and survival, and their sensitivity to atmospheric parameters, not only was this primary intent effectively accomplished, but it was found possible to extrapolate the resulting design to include several scientific experiments without substantially violating any of the basic tradeoffs which were made en route.

The greatest value of such an "Evaluation of Fundamentals" study seems to lie in its application to a new realm of missions. We are currently applying it to our first-in-house study of Jupiter flyby missions.

Outputs Desired from Conceptual Designs

A conceptual design study should be accomplished with sufficient thoroughness to satisfy the following technical requirements:

(1) System engineering should be carried to the point where there is a reasonable assurance that all subsystem interactions have been exposed.
(2) At each tradeoff decision point, an analysis should be provided of the alternatives considered and how the decision was made.
(3) At each tradeoff decision point, one specific selection should be made from the possible alternatives and carried through as a 'baseline' concept.
(4) One 'baseline' concept should be evaluated to the point where there is either a reasonable substantiation of feasibility or the exceptions are emphasized as potential problem areas.

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*See JPL Engineering Planning Document 261, "Mariner Mars 1969 Lander Technical Feasibility Study," by R. W. Davies.

(5) The 'baseline' concept need not be an optimized system; alternative approaches which appear to be feasible should be listed for possible use in optimization after project approval.

(6) Potential problem areas should be summarized, listing their criticalness to the mission, and wherever possible, suggesting alternate approaches for implementation into the research and development programs.

(7) If more than one mission is involved in the study, efforts should concentrate on the earliest mission, with a less-rigorous analysis of the growth capabilities included.

In addition to the generation of these technical characteristics, the study contractor is typically asked to also present estimates of schedules, funding, manpower, and resource requirements.

GROSS PROJECT PLAN

When the technical evaluation of a potential mission has been completed and if the results indicate that the mission is both desirable and feasible, then a gross project plan is prepared to evaluate its potential for implementation as a project.

The plan at this stage is quite informal but is generated in a specific format. The key elements of such a plan are shown in Figure 6.

First, the objectives, both scientific and engineering, are defined. The value of the proposed mission is compared to other possible missions in order to determine its justification. It is also necessary to justify the basis on which the evaluation is made, in other words, to summarize the amount of study accomplished and its thoroughness.

The main body of the proposal is a description of the technical plan. This includes an outline of the nature of the project, including all systems and principal subsystems, as well as the flight missions and, to the extent known, the technical design parameters. The technical approach is defined by an evaluation of the technical problems expected against the current state-of-the-art, showing where advances in the state-of-the-art are required and how these advances might best be approached.

A management plan is generated, showing possible assignments to various NASA centers, suggested management organization, and relationship with non-NASA agencies.

A proposed schedule is listed, showing flight dates, and key milestones for Phases B, C and D.

A procurement plan is generated, suggesting how much of the project should be contracted to industry, and how and when such procurement should take place.

Estimates of resource requirements are an important aspect of the proposal. Based on the procurement and management plans, the internal NASA manpower requirements are listed. Based on the technical plan, a list of major facility requirements is generated. Based on all these requirements, an estimate of the funding requirements, year by year, is then generated.

Finally, if there are any unique aspects to the public release of scientific results, such as complex processing requirements, a plan is generated showing how this might be accomplished.

SELECTING A PROJECT FOR IMPLEMENTATION

As more and more potential missions pass through the conceptual design phase, the accumulation of project plans of course increase. Even just the passage of time can result in modifications to a proposal for such reasons as revised launch dates, advances in state-of-the-art, revisions in our knowledge of planet environments, improvements in the technical plan, and many others. In the case of the Voyager project alone, during the three years of its Phase A activities, approximately eight separate project plans were processed before all conditions merged to produce Phase B go-ahead.

Figure 7 is a functional portrayal of this final activity of a typical Project Selection phase. The criteria which probably have the greatest influence on the selection of the next go-ahead plan are scientific desirability, the technical plan, economics, and politics.

The scientific desirability is based on expression, by the scientific community, of <u>relative</u> interests in the various missions which can be initiated. Such an expression is very difficult for the project planner to obtain - or,

perhaps more correctly, it is very difficult for the project planner to obtain a single expression of relative scientific priorities which represents the views of more than one small segment of the scientific community.

In cases where common agreement can be reached, a significant impetus can be generated in the project selection process. The letter from the National Academy of Sciences recommending that the search for life on Mars be established as the next significant objective of our space program, played an important role in establishing the Voyager project and undoubtedly will continue to affect future decision milestones in that project.

The technical plan is important in that it defines the sequence of missions to be performed, and indicates an evolution of experiments. It expresses the risks involved in a particular mission and allows an evaluation of the "efficiency" of a particular mission, i.e., the probable scientific value to be obtained as a function of the resources expended.

The economics of the situation places a ceiling on the number of active projects within NASA at any given time. The total NASA allocation appears to have stabilized at just over five billion dollars a year. When approved commitments in flight projects such as Gemini, Apollo, Surveyor, etc., and in supporting programs such as Advanced Research and Technology, tracking and data aquisition, etc., are projected into future years, an estimate can then be made of the probable funds available for new programs. The effect on various candidate projects can then be evaluated.

The word "politics" as used in Figure 7 is assigned a somewhat broader definition than it usually has. In figure 7, it includes, in addition to congressional action, such deliberations as redistribution of emphasis on the various programs within NASA, redistribution of resources, and in general, is intended to encompass all those major forces affecting project selection which are difficult to evaluate and predict from the vantage point of a technical man in a planning office within a NASA center.

By applying the above criteria to the available project plans, the long range plan is modified if necessary as a result of this new information, and this plan is then used as a guide for all affected personnel.

However, it should be pointed out that the combined effects of the above four criteria are variable with time as conditions change. This final process of the project selection is hence a continuous affair. The long range plan is consequently also varied frequently. During the last few years, the long range plan for the unmanned planetary program has been modified significantly each year and our present indications are that modification will continue for some time to come.

On those occasions when a particular project appears headed for implementation, as was the case during November, 1964, with the Voyager, the NASA Headquarters program manager and the cognizant NASA center's project manager are appointed and authorized to begin the staffing of their organizations. One of the first significant activities then accomplished is the conversion of an informal project proposal into a formal Preliminary Project Development Plan which forms the basis for formal Phase B approval.

CURRENT CANDIDATES FOR PLANETARY EXPLORATION

Now that the criteria and functions of the Project Selection process have been defined, it seems appropriate to describe the current status of project planning in the unmanned planetary program (which encompasses planets, their moons, asteroids, and comets).

With the completion of the Mariner Venus and Mariner Mars missions, the unmanned planetary program currently consists of only the Voyager Mars project. The Voyager consists of two major modules: an orbiter bus and a lander. The orbiter bus is nearing completion of its Phase B activities; the lander is entering Phase B. We are assuming that the orbiter bus and lander schedules will eventually merge and that the combined system will be able to successfully pass all requirements for entry into the operations phase. We assume further that the Voyager concept will be capable of exploring Mars for a decade or more.

The logical objective, therefore, of planning should be the determination of the optimum mix of missions to the other bodies of the solar system. In this regard, there could be three classes of candidates: Voyagers, Mariners, and what might be termed Advanced Planetary Probes.

The Voyager class of missions could be defined as an orbiter and/or a lander which are capable of a comprehensive, long-term exploration of perhaps all planets in the solar system. The scientific objectives could be roughly comparable to those of Voyager Mars. In terms of scientific payloads, the Voyager spacecraft probably represents two or three hundred pounds of scientific equipment in each of the orbiter bus and the lander. Some of the other spacecraft equipment, however, might be somewhat different from the Mars version. For example, on flights to Jupiter and beyond, solar cells are likely not the optimum power source and probably would be replaced by a nuclear or chemical source. On distant missions either communication data rates would be reduced or more powerful transmitters would need to be incorporated. The cost of a Voyager-type mission is probably the highest of the three candidate classes.

The Mariner class of missions could be defined as a flyby mission with perhaps a small atmospheric probe or a minimum survival capsule. The flyby objectives could be roughly comparable to those of the Mariner Mars but perhaps with some replacement of instruments depending on the target. It would be applicable to missions to some of the nearer solar system bodies, such as Venus, perhaps Mercury, comets and near asteroids. The cost of such a mission is probably significantly less than that of the Voyager.

The Advanced Planetary Probe is a fairly recent concept which is just now entering the Parametric Evaluation phase. Its primary characteristic is that it is by far the least expensive of the three candidates. It is envisioned as being a simple flyby mission to each of the outer planets and serving as a precursor probe to later Voyager missions. It is hoped that its scientific objectives will permit at least a few fields and particles experiments. The feasibility of such a concept is yet to be determined.

In considering potential schedules for the Voyager class of missions, the determining factors to be balanced against scientific desirability appear to be the high cost, the state-of-the-art, lead time and modifications required to the basic design. On the assumption that substantial funds will not be available for other projects until the peak of the Voyager Mars funding has been passed, the initiation of a Voyager-to-other-planets could not occur before 1971 or 1972. However, it makes sense to complete the development and testing of the base-line Mars version before attempting to modify it for other applications anyway, and this factor also leads to a no-earlier than 1971 or 1972 start. The lead time would probably be on the order of three to five years depending on the magnitude of modifications required. Within these assumptions, the first application of the Voyager concept to a planet other than Mars cannot be expected before mid- to late-1970's. The modifications required for a Venus mission are probably less than for any other planet. Hence a Voyager Venus mission could be implemented before that of any other planet. In a gross sense, then, if the scientific desires are compatible, we can envision a substantial program of exploration of Venus in the latter half of the 1970's and early 1980's, and of the other planets in the 1980's and 1990's.

We have studied the potential applications of a Mariner type of mission to comets and near asteroids and find that a Mariner Mars type of spacecraft is almost directly applicable. It should therefore be possible to implement such a mission before the Voyagers described above. If a second generation Venus flyby mission is desired, which is not greatly more complex than the Mariner class, a launch could probably be made on about the same time scale as the comet mission. The requirements of a Mercury mission are not too well defined yet, but a general review indicates that a mid-1970's mission could probably be accomplished.

If the feasibility of the Advanced Planetary Probe can be confirmed with characteristics similar to the definition provided earlier, it should be possible to begin probing the asteroid belt and beyond within the next few years.

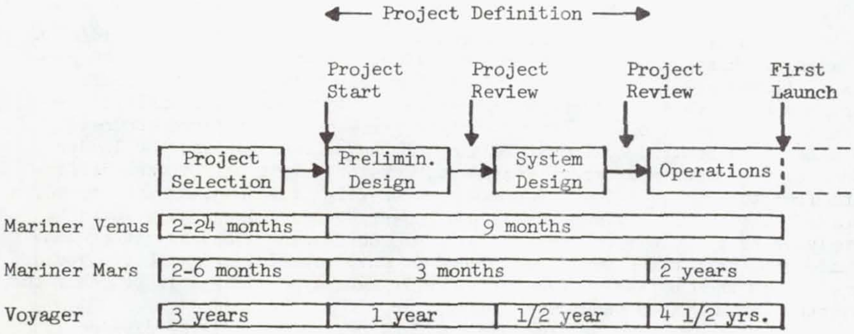
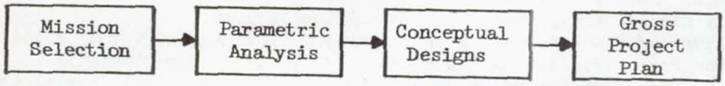
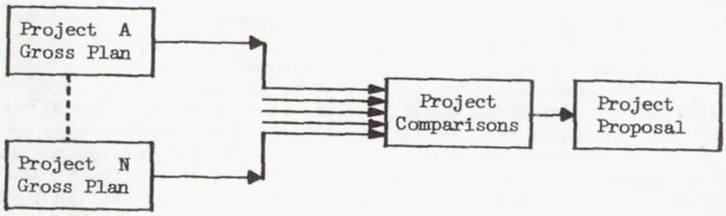


FIGURE 1

Typical Project Phases and Time Scales



a. Preparation of each project plan



b. Comparison of project plans

FIGURE 2

Major Elements of Typical Project Selection Phase

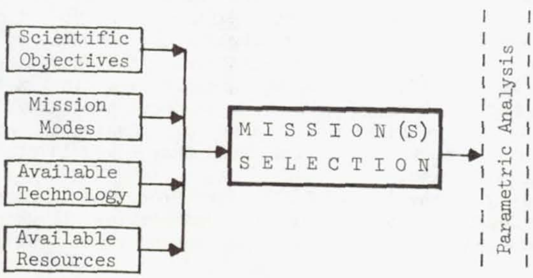


FIGURE 3

The Mission Selection Process

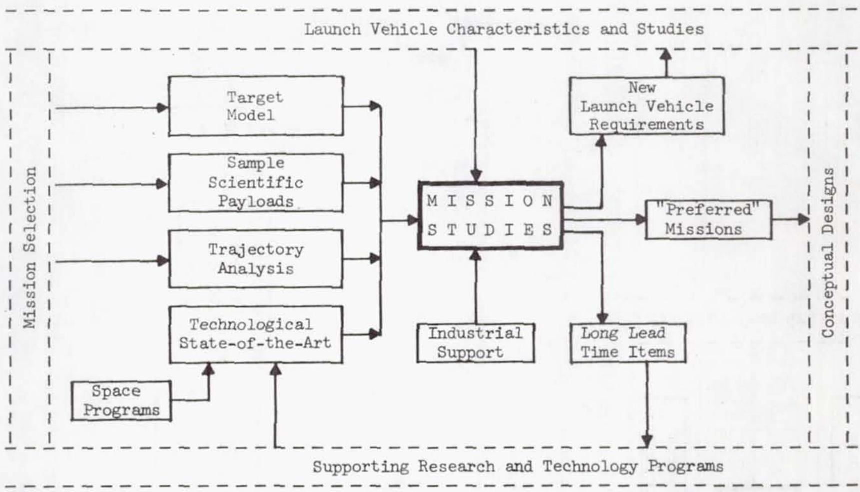


FIGURE 4

The Parametric Analysis Process

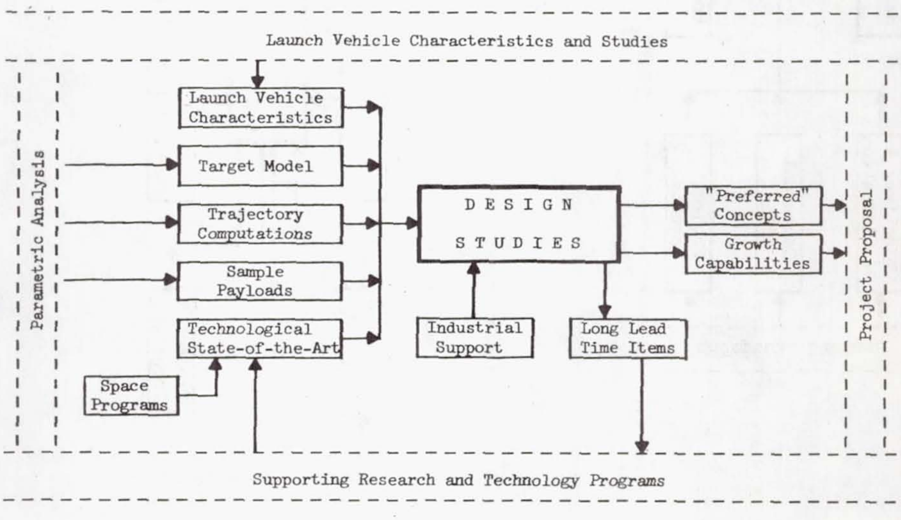


FIGURE 5

The Conceptual Design Process

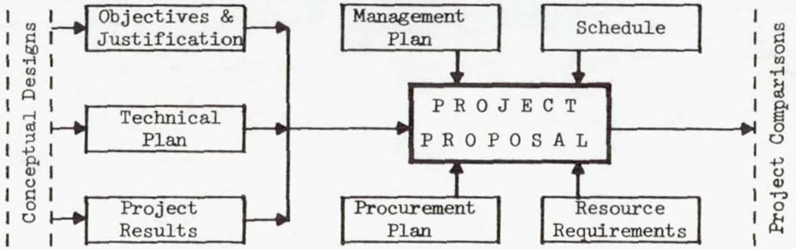


FIGURE 6

Preparation of the Gross Project Plan

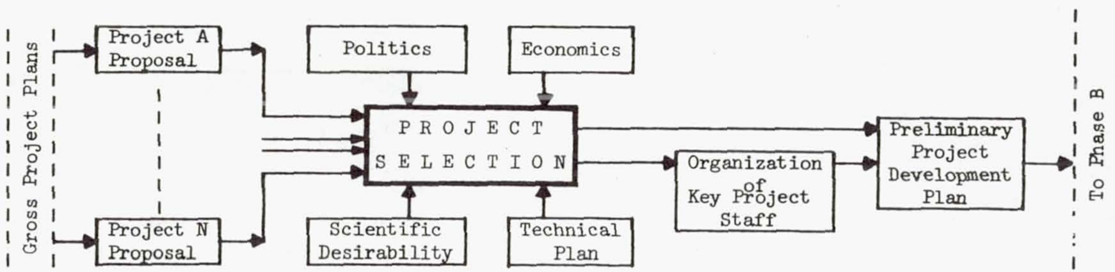


FIGURE 7

Completing the Project Selection Phase

ORBITER MISSION DESIGN PROBLEMS

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INTRODUCTION

With the establishment of the Voyager program, NASA is planning to perform orbiting and landing missions at Mars, beginning with the 1971 opportunity. I want to examine, in considerable detail, one particular facet of orbiter missions, that of planetary orbit design and selection. Where numerical examples are cited, the 1969 Mars opportunity will be employed. I will use this particular case for two reasons. First, I am associated with a group at JPL which recently finished a study of a 1969 Mars orbiter; thus the data is readily available. Second, there are no current plans to perform a Mars orbiter mission in 1969; as a result, the material and opinions I present are my own, and do not necessarily reflect the views of the Voyager project personnel.

When compared with an orbiter mission, the choice of the Mariner II and Mariner IV flyby trajectories was a fairly simple straightforward matter. (Such an assertion is, of course, somewhat easier to defend when standing in the shadow of success.) Let me explain what I mean.

At the time of the Mariner II flight to Venus, we had never before performed a midcourse trajectory correction maneuver. We were pretty green at the game of flight path mechanics (and of building spacecraft, for that matter)! Our estimates of orbit determination capability and of midcourse maneuver execution errors were rather high. The general objective was to fly close enough to the planet to obtain reasonable resolution with the infra-red experiment and to make magnetic field strength measurements. This desire had to compete with firm requirements not to impact the planet or to occult the Sun or the Earth. Without solar illumination the spacecraft could not generate sufficient power for the planetary encounter mode; without sight of the Earth the spacecraft would lose its third axis attitude reference and could no longer point its high-gain antenna toward Earth.

The a priori estimate of the semi-major axis of the trajectory dispersion ellipse was as much as 6000 km (1σ). There was no requirement for precise control of time-of-flight except to assure that the encounter sequence occurred over the DSIF tracking station at Goldstone, California.

Mariner IV set out with somewhat more ambitious objectives; the ability to perform two midcourse trajectory corrections was provided, although the second maneuver was originally only intended as a "backup" to the first. The a priori estimates of trajectory dispersions were down to about 4600 km (1σ) due to analysis of the Mariner II tracking data plus improvements in the stability of the DSIF radio equipment. This meant that the experimenters (especially the TV team) could be fairly specific in selecting a single interesting target latitude and longitude, the latter to be achieved by controlling the time-of-flight; the pre-flight accuracy estimate was 2100 sec (1σ).

The situation became a bit more complicated when RF occultation came along as an experiment late in the game, after the spacecraft design had been frozen. The flyby aiming point had to be moved a bit, and the acceptable target zone reduced in area, to <u>assure</u> occultation of the Earth but <u>not</u> of the Sun, and still pass over a reasonably interesting area to photograph.

In both the Mariner II and Mariner IV missions there was only a single flyby pass of the target and only a single set of encounter data. The encounter experiments were performed only once, and the experiments were completed within a few minutes time.

Finally, both Mariner projects were blessed, I believe, by being severely weight-constrained. There is a certain advantage in being weight-limited

because it limits what you are asked to attempt to do; hence, you can concentrate your efforts on doing fewer objectives well.

Contemplated orbiter missions are a bit different, however, for at least three reasons. First, the ability to control the flight path will be considerably improved when final reduction of the Mariner IV tracking data has been completed. Estimates of 50 to 150 km (1σ) dispersion appear reasonable, using two midcourse maneuvers based upon Earth-based radio tracking. Second, the experiments to be performed in orbit will be repeated again and again, with ever changing orbit geometry, illumination, and target viewing angles. Third, if the present Voyager launching vehicle expectations are maintained, we will send orbiting spacecraft to Mars weighing in excess of 5000 lb. Over 2000 lb of this will be active spacecraft weight in orbit, the difference being used for propellant to establish the orbit. With a capability for over 2000 lb of spacecraft, the number and variety of experiments which can be attempted, and the conflicting requirements which will result, will be great indeed. The ability, and hence the pressure, to attempt more ambitious objectives will significantly complicate the planning, design, and esecution of the mission.

With the preceeding discussion in mind, let us consider in detail the problem of orbit design and selection for a hypothetical orbiting mission.

EARTH-MARS TRAJECTORIES

Basic Trajectory Characteristics

In considering a Mars orbiter mission, care must be taken to distinguish between two kinds of trajectories. One is the interplanetary trajectory and is associated with the transfer of the spacecraft from Earth to Mars. The other trajectory is that associated with the motion of the orbiter in a satellite orbit about Mars. The latter is often highly constrained by the former. Thus I shall introduce the problem by consideration of the interplanetary trajectory and its selection.

To begin with, we are constrained by the payload capability of the intended launching vehicle. The relationship between Earth departure energy and payload for a typical vehicle is shown in Figure 1. The vis-viva energy, C_3, is twice the geocentric energy per unit mass. The higher the energy required to reach a given target planet, the lower the available payload.

Figure 2 illustrates the energy requirements for one way ballistic trajectories to Mars for the 1969 opportunity. The abscissa and ordinate are Earth launch date and Mars arrival date, respectively. Contours of constant departure energy are shown with values of C_3 running from 8 to 30 km^2/sec^2. Also included are lines of constant flight time, T_f, as well as a scale showing the Earth-Mars distance directly affects the communication system capability.

There are two major types of interplanetary trajectories: Type I being defined as those for which the heliocentric central transfer angle from launch to encounter is less than 180 degrees, and Type II those for which the transfer angle is greater than 180 degrees. Type I trajectories are represented by those contours in the lower right and Type II by those in the upper left region. As an example, suppose we wanted to launch a 1940 lb spacecraft to Mars. From Figure 1 this requires a C_3 of 10 km^2/sec^2. Figure 2 shows that we could launch such a payload on a Type I trajectory any day between February 18 and March 16, 1969. If a longer period were desired for launch operations, we could achieve this by increasing the injection energy at the expense of spacecraft weight. Thus an 1800 lb payload could be launched with a C_3 of 12 km^2/sec^2 during the period February 10 to March 25.

Another factor to be considered in selecting the transfer trajectory is the approach velocity at the target. Figure 3 presents curves of constant hyperbolic excess velocity at Mars (VPL). As VPL increases, so does the percentage of spacecraft mass which must be used as propellant to place the payload into a given Mars orbit.

Curves of constant orbiter mass are derived from tradeoffs between C_3 and VPL; the higher C_3 becomes, the lower VPL must become in order to achieve the same orbiter mass. Figure 4 shows curves of constant optimum orbiter mass for 20, 30, and 50-day firing periods assuming continuous variable propellant loading.

Figure 4 also illustrates another parameter which must be considered in the selection of Earth-Mars transfer trajectories, namely, the declination of

the outgoing geocentric asymptote (DLA). In past lunar and planetary missions, launch azimuths between 90 deg (due east) and 114 deg (east-southeast) have been used. However, when the absolute magnitude of the declination of the outgoing asymptote becomes greater than the latitude of the launch site (28.3 deg for Cape Kennedy), there exists a band of launch azimuths symmetric about 90 deg in which it is not possible to launch without prohibitive dog-leg maneuvers. From Figure 5, which illustrates the size of this band, we see that for declinations greater in absolute magnitude than 36 deg it would not be possible to use launch azimuths between 66 deg and 114 deg; hence the "traditional" launch azimuths could not be used.

The question of which launch azimuths can be used depends not only upon the required tracking and telemetry coverage but also upon the probability of the vehicle impacting an inhabited area due to a malfunction during boost. Allowable launch azimuths are specified by the range safety group at the Eastern Test Range, and involve a tradeoff between the mission requirements for a given launch azimuth and the impact probability for that particular azimuth. Figure 6 shows a typical kill probability estimate as a function of launch azimuth for the Eastern Test Range (ETR).

Mars Arrival Conditions

The heliocentric locations of the Earth and Mars for the Type I and Type II range of arrival dates are shown in Figure 7. It is interesting to note that the Earth-Mars-Sun angle for all arrivals is about 45 to 46 degrees, thus simplifying the mechanization of the spacecraft's high-gain antenna. (It was this phenomena which permitted the Mariner IV spacecraft to employ a body-fixed high-gain antenna for the recent Mars encounter.) Figure 8 shows the behavior of Earth-Mars distance and the Earth-Mars-Sun angle as a function of calendar date.

The trajectory data presented thus far is summarized in Figure 9. In addition to the terminology already introduced, the communication distance, CD, is shown at Mars encounter (E), and 180 days following encounter (E + 180). Similarly, the solar distance, SD, is also shown. The Mars heliocentric longitude was illustrated previously (see Figure 7). In general, Type I trajectories are seen to have shorter flight times than Type II's (185 vs. 270 days), arrive at least a month earlier, and thus have lower communication distances at encounter (140×10^6 vs. 170×10^6 km).

SATELLITE ORBIT SELECTION

Let us turn our attention now to the Mars encounter itself and the selection of Mars satellite orbits.

The primary justification for the objective of long-life Mars orbiter missions is the desire to observe seasonal changes at the planet. Minimum mission lifetimes of 3 months to 6 months have been suggested; some have suggested missions for an entire Martian year (22 1/2 Earth months). One might then logically ask when and where the major surface changes or "wave of darkening" occurs. In his book Physics of the Planet Mars[1], De Vaucouleurs makes the following remarks:

> *"By marking the heliocentric longitude η_o at which the upwards trend of the variation curves becomes noticeable (at some arbitrary level above the minimum intensity), i.e., by plotting the points of incipient darkening, it is even possible to trace with fair accuracy the propagation of the 'front' of a darkening wave progressing evenly northward from the south polar area.*
>
> *"This wave starts near the end of the southern winter (η_o = 250 degrees) at about latitude ϕ = -60 degrees; it then spreads and crosses the equator before mid-spring (η_o = 290 to 300 degrees), reaching latitude +40 degrees before the end of the southern spring (η_o = 330 degrees). The front of the wave thus covers 100 degrees in latitude (6000 km) in about 130 days, travelling at some 45 km per day (or 0.5 m per sec). There is a possibility that the prepagation might not be exactly synchronous at all longitudes around the glove, but the observational material is not yet sufficient to confirm this."*

Figure 10 has been extracted from this same reference and illustrates the northward progression of the wave of darkening as a function of the heliocentric longitude of Mars. This figure would seem to suggest the desirability of arriving earlier in the Martian year (than the 1969 Type I trajectories permit) when the rate of increasing darkening is a maximum in the southern hemisphere, in order to see the greatest difference in darkness intensity levels during a given orbit lifetime following encounter. It should be noted, however, that other interpretations of the preferred arrival time have been proposed (by Sagan and others) which suggest that the previously described 1969 arrival dates are attractive. Most of these points of view are based upon a figure presented in a 1962 paper by J. H. Focas[2]. Perhaps it is felt that if the orbit lifetime is as long as six months, the darkening will have diminished in certain areas so that the observation of a change might be possible.

Before placing a requirement on the orbit lifetime of 3 to 6 months, however, the whole question involving the likelihood of "observing" a seasonal change should be carefully investigated. Such questions should be answered as: (a) is the optimum arrival period at the time of maximum rate of increasing darkening, or after the level of greatest darkening has occurred, (b) what is the likelihood of observing the same area twice at a suitable interval, and (c) even if a given area is observed twice, does the probability of detecting a change depend upon also having similar lighting conditions and viewing angles for both observations?

General Orbit Considerations

A large number of considerations are involved in attempting to select the optimum parameter values for a satellite orbit about Mars. I will describe the most important parameters and indicate those factors which most strongly influence their selection. During the discussion, it will be noted that one of the most significant constraints in designing an orbiter mission about Mars involves the selection of a periapsis altitude which is felt to be of sufficient height to ensure a probability of 10^{-4} that an unsterile orbiter will not decay into the Mars atmosphere prior to contamination by some later mission. Depending upon the assumed high-altitude density model, values of periapsis altitude from about 1000 km to 5000 km would be necessary to provide a lifetime of the order of 50 years.

Figure 11 illustrates the basic parameters that are customarily used to define the shape and orientation of a planetary satellite orbit. The eccentricity and semi-major axis are denoted by e and a, respectively. It should be noted that periapsis altitude has been shown rather than periapsis radius or distance from the planet center. This has been done because such considerations as drag effects upon orbit lifetime, instrument height above the surface, and photographic resolution make altitude a more useful parameter than distance from the planet center. In Figure 11 the assumed value for the Mars oblateness coefficient, J, is 0.00292. Because of this oblateness the satellite orbit does not remain fixed inertially. The approximate secular rates of motion are given in the figure.

Now let us examine how we establish such a satellite orbit. The optimum place to establish an orbit about Mars is at periapsis of the approach trajectory hyperbola, and the optimum thrust direction is parallel to the periapsis velocity vector. The encounter geometry is shown in Figure 12. The actual "aiming point" is the intersection of the incoming trajectory asymptote with the $\overline{R}$-$\overline{T}$ plane defined in the figure. This point is usually specified in polar coordinates, with the radial distance denoted by $\overline{B}$ (the impact parameter) and the angle, measured clockwise from $\overline{T}$ in the $\overline{R}$-$\overline{T}$ plane when looking along $\overline{S}$, denoted by θ. Note that the actual point of closest approach, periapsis of the flyby trajectory, is closer to the planet than the $\overline{B}$ vector; furthermore, as $\overline{B}$ is reduced, the inward bending of the trajectory becomes even more pronounced.

Figure 13 shows a more generalized situation; it can be seen that a variety of aiming points might be considered which result in different orbit inclinations with respect to the Mars equator, different latitudes for the location of the sub-periapsis point, different lighting conditions near periapsis, different orbit precession rates due to Mars oblateness, different occultation times for other bodies, and several other varying characteristics. In the absence of yaw or out-of-plane maneuvers in performing the

orbit insertion, the family of all possible satellite orbit planes contains the approach asymptote vector S.

Orbit Selection Constraints

We shall consider certain orbit selection constraints in turn. The following conditions are typical of those which might be desirable for a particular mission:

(1) It is desirable that Sun occultation be avoided for as long as possible.
(2) If a useful Earth occultation experiment is to be performed, it may be desirable that the Earth be occulted during each orbit for several days (up to perhaps a month) following initial orbit establishment, but the occultation time per orbit must not be so large as to significantly reduce that portion of the orbit period needed for data transmission.
(3) It is desirable that the angle between Canopus and the near planet limb (as measured at the spacecraft) remain above some prespecified value (perhaps 20 to 50 degrees, depending upon the control policy and field-of-view requirements of the Canopus tracker).
(4) During periods of photographic observation it is desirable that the following lighting and planetary coverage conditions exist simultaneously somewhere "reasonably" near periapsis:

    (a) The Sun-orbiter-planet angle should lie between about 100 and 140 deg.
    (b) The orbiter should be at a Mars latitude of between about 40 deg south and 10 deg north.
    (c) The orbiter altitude should preferably not exceed about 2 to 3 times the periapsis altitude. The last condition is met as long as the orbiter true anomaly (in-plane angle from periapsis) is within about 60 to 80 degrees from periapsis (altitude 2 x H_p), or about 80 to 100 degrees from periapsis (altitude 3 x H_p).

(5) It is desirable that nodal and apsidal precession rates due to Mars oblateness tend to improve as many of the above conditions as reasonably possible.
(6) There will probably be a required orbit lifetime of at least some minimum number of years to satisfy the non-contamination constraint. The actual orbit lifetime will depend primarily upon estimates of atmospheric drag and secondarily upon third-body gravitational effects and other possible perturbations.

Periapsis Altitude Considerations

The selection of values for the critical orbit parameters is influenced by many considerations. For example, the following factors govern the choice of a nominal periapsis altitude:

(1) The strongest factor is probably the required orbit lifetime which depends primarily upon the high-altitude density profile and secondarily upon the apoapsis altitude. Another factor which must be considered in the prediction of orbit lifetime is the long-term effect of the Sun (third-body influence) upon the orbit periapsis altitude. Preliminary considerations indicate that third-body effects can cause the periapsis altitude to diminish by as much as 50 to 200 km during a 50 year period for orbits with large semimajor axes.
(2) Another important factor which influences the choice of periapsis altitude is the estimated dispersion in this parameter due to orbit determination errors, midcourse guidance errors, and orbit insertion guidance errors. The importance of obtaining accurate estimates for those errors increases with the desire to achieve as low a periapsis altitude as possible consistent with the orbit lifetime requirements. Typical estimates for the combined 3 σ error in periapsis altitude due to the above sources are in the range of 500 to 1,000 km.

(3) For a given apoapsis altitude, the in-orbit payload capability can be increased by lowering the periapsis altitude. This can become very important whenever the mission performance capability is marginal.

(4) The scientific desire to orbit close to the Mars surface in order to obtain high resolution coverage obviously affects the selection of the nominal periapsis altitude. We must answer such questions as: (a) do we want to get as close as guidance accuracy permits? (b) what percentage error in controlling periapsis altitude is acceptable? and (c) for small H_p and large H_a, does the relatively high periapsis velocity cause image motion problems?

(5) Any consideration of orbit trim affects the selection of both the initial and final periapsis altitudes. When mission orbiter weight capability is not marginal and we can afford the luxury of additional retro fuel, an orbit-trim adjustment policy offers certain advantages. By initially choosing a conservative periapsis altitude, the danger of impact is minimized. Further, after determining the initial orbital elements accurately, an orbit trim maneuver can be utilized to achieve better control of the final periapsis altitude. On the other hand, orbit-trim maneuvers must be very carefully implemented, because failure to shut-off such a maneuver might result in planetary impact.

(6) For a given orbit inclination and apoapsis, the periapsis altitude can be selected so as to increase the chances that certain bodies will or will not be occulted to the orbiter by Mars. Lowering H_p improves the chances of ensuring Earth occultation but, at the same time, brings Canopus closer to the limb of Mars.

Apoapsis Altitude Considerations

The following factors influence the selection of an acceptable apoapsis altitude:

(1) In-orbit payload requirements may require the use of a large apoapsis altitude. The effect of apoapsis altitude upon in-orbit spacecraft mass is shown in Figure 14.

(2) Orbit sensitivity (variations in apoapsis altitude, orbit period, etc.) to orbit insertion errors increases as the nominal apoapsis altitude is raised. For the guidance accuracies previously quoted, nominal apoapsis altitudes should not exceed 50,000 km to assure capture.

(3) For a given orbit inclination and periapsis altitude, the selection of a high apoapsis altitude reduces the likelihood of occultation of certain bodies. Also, a higher value for H_a permits a slightly lower value of H_p to be chosen for the same orbit lifetime in the presence of atmospheric drag.

(4) The orbit period obviously increases with higher values of H_a; this gives ground facilities more time to prepare for each major scientific data collection sequence near periapsis. This may well be a significant consideration.

(5) Orbit precession rates (nodal regression Ω and apsidal precession ω) due to Mars oblateness depend strongly upon the semi-major axis and therefore upon the apoapsis altitude. With careful planning, this orbit precession can usually be used to advantage by either causing the sub-periapsis point to move into more favorable latitude regions, by trying to delay Sun occultation, or by trying to reduce certain third-body effects. With very high apoapsis altitudes, however, the precession rates are too low to be of any significant value. For apoapsis altitudes above about 20,000 km the precession rates are less than the apparent motion of the Sun of about 0.5 deg/day (due to Mars' motion about the Sun).

(6) If it were possible to achieve the exact nominal H_p and H_a values, it would be possible to select an orbital period which results in evenly mapping the planet surface as well as returning to the same area after a given period of time such as one month, for example, in order to increase the chances of observing a seasonal change.

In general, guidance errors preclude the accurate establishment of such a pre-selected orbit period.

Orbit Inclination Considerations

(1) From the standpoint of total planet surface coverage it is desirable to use as high inclination orbits as possible. However, for early missions it may be desirable to use lower inclination orbits (less than 45 deg) for the following reasons. First of all, given that only a reasonable number of pictures can be returned, these should concentrate on the most interesting surface-feature regions; at least prior to Mariner IV these areas were presumed to lie within 30 or 40 deg of the Mars equator (to the south). By using orbit inclinations close to these latitude regions, longer dwell times over these preferred latitudes can be achieved, thus increasing the chances of passing over a given area twice during the mission lifetime in order to observe possible seasonal changes.

(2) The choice of orbit inclination obviously affects the occultation geometry with other bodies such as the Sun, Earth, and Canopus. Low inclination orbits increase the chances of occulting the former two bodies but leaving Canopus well in view, while high inclination orbits tend to have the opposite effect. We have to select low inclination orbits carefully so that Earth occultation will be achieved for some period of time but without Sun occultation. Also, it must be remembered that, because the Mars equator is inclined some 25 degrees to the ecliptic plane, orbits having equal north and south inclinations to the Mars equator have quite different inclinations to the ecliptic plane, and it is this latter reference plane which more appropriately governs the geometry of occultation of the Sun, Earth, and Canopus.

(3) As we noted previously the orbit inclination affects the nodal and apsidal precession rates due to Mars oblateness. To first order, there is no movement of the line of apsides within the orbit plane for $I = 63.4$ deg. For $I>63.4$ deg, apsidal movement is opposite to that of the orbiter and, for $I<63.4$ deg, apsidal movement is in the same direction as the orbiter. The nodal regression rate is zero for $I = 90$ deg and increases as I decreases. These effects for a typical Mars orbit are shown in Figures 15 and 16. It should be remembered, of course, that these orbit effects are strongly dependent on the orbit's eccentricity. Increasing the periapsis altitude from 1500 km to 4000 km would reduce these precession rates by more than an order of magnitude.

(4) The ability to accurately determine the orbit of a Mars satellite by Earth-based radio doppler measurements is a strong function of the geometry of the orbit, as viewed from Earth. At the present time, it appears that the only constraints which this factor might place upon the choice of the satellite orbit are: (a) the angular rate of the orbit plane with respect to the Earth-planet line-of-sight should be non-zero, (b) the orbit inclination with respect to the plane generated by the Earth-planet line-of-sight should be non-zero (this latter plane is very close to the ecliptic plane), and (c) the orbit plane should not be perpendicular to the Earth-Mars line. There is a very low probability that any of these situations would occur exactly, and slight departures from the above situations generally allow satisfactory orbit determination.

Periapsis Location Consideration

The location of periapsis is important for very elliptical orbits and is influenced by such considerations as:

(1) It is desirable that periapsis occur on the Earth-side of Mars so that the entire orbit insertion maneuver can be observed from Earth. The Type I and Type II trajectories previously described for the 1969 Mars opportunity satisfy this condition.

(2) The location of the sub-periapsis point should preferably occur in the Mars latitude band between about 40 degrees south to 10 degrees north. In order to avoid orbits which exhibit Sun occultation,

however, aiming points usually have to be chosen which result in initial periapsis latitudes slightly outside of this desired band. Selection of orbits having $I<63.4$ deg will produce a movement of the sub-periapsis point towards the desired region with time. One must, however, trade off the sub-periapsis latitude location against the desired lighting conditions beneath the orbiter.

(3) As stated previously, it is most efficient to have the orbit periapsis coincide with the natural periapsis of the approach hyperbola. In an actual case, however, either the orbit insertion guidance mode employed or errors in the insertion maneuver can result in rotating this apsidal line by several degrees in the plane of motion. With extra retro fuel, one could consider an intended rotation of the apsidal line for the purpose of initially locating the sub-periapsis point over some preferred region. For a marginal mission (from a weight viewpoint), however, rotations in excess of about 10 to 20 degrees are probably not feasible.

Approach Geometry

Typical approach aiming diagrams for Type I and Type II trajectories are shown in Figures 17 and 18. These figures represent a "bullseye" view of the target plane as seen from the spacecraft when it is at a great distance from Mars. The contours shown designate those regions in the B-vector space which would cause a flyby spacecraft to be occulted from a particular body. The symbols EPM, SPM, and CPM refer to the Earth-probe-Mars, Sun-probe-Mars, and Canopus-probe-Mars angles, respectively. As can be seen from these figures, the Type I trajectories approach Mars from the lighted side while the Type II trajectories approach from slightly on the dark side. (This is characteristic of Type I and Type II trajectories for any Mars opportunity.) The Type I approach asympotes are inclined some 9 to 14 deg to the ecliptic plane (coming from below), while the Type II trajectories approach Mars from 15 to 26 deg above the ecliptic plane.

Because the Type I trajectories approach Mars from the lighted side (relative motion with respect to Mars), aiming points selected on this side result in direct motion of the orbiter about Mars and have periapsis on the lighted side but initially closer to the evening terminator. Since the Type II trajectories approach from slightly on the dark side of the morning terminator, direct orbits result in periapsis being located about halfway between the two terminators. After a period of a few weeks, however, the motion of Mars about the Sun moves the morning terminator closer to the sub-periapsis point.

Orbit Lifetime Results

Several factors can cause the periapsis altitude of a satellite orbit to vary:

(a) atmosphere drag
(b) solar gravitation
(c) planet oblateness
(d) solar radiation pressure

When the satellite is at low altitudes, the dominant effect is atmospheric drag. Although estimates of the near-surface atmospheric densities are believed known to a factor of perhaps two, the high altitude densities are uncertain by several orders of magnitude. In order to demonstrate the effect of these large uncertainties, orbit lifetimes have been computed for a variety of atmospheric models. Model I was generated at JPL and corresponds to a more or less nominal density profile, whereas Model II was generated by the NASA Mars Standard Atmospheric Committee[3] and corresponds to a "maximum" density profile. (See Figure 19.) Now, in all fairness, it should be noted that this latter model was generated for entirely different purposes (namely, entry body studies) and was not intended for use in satellite drag computations; it assumes a constant adiabatic lapse rate of 2° K/km to an altitude of 2000 km and above. Nevertheless, it is instructive to see the effect which such a model produces.

The orbit lifetime can be determined approximately by computing the time required for an elliptical orbit to decay into a circular orbit and then

computing the additional time required for the circular orbit to decay to the Mars surface. In addition to the atmospheric density, the orbit lifetime also depends upon the ballistic coefficient of the orbiter. The ballistic coefficient is defined by $K = m/C_D A$, where C_D is the drag coefficient (assumed equal to 2 for the high altitude regime), A is the effective area, and m is the orbit mass. Orbit lifetime varies directly with the ballistic coefficient, and doubling K doubles the orbit lifetime, etc. The ballistic coefficient will, of course, depend upon the cross-section which the orbiter presents throughout the orbit. A typical long-term "effective" value of K for a solar-powered orbiter might be approximately 0.15 slugs/ft^2.

The results of such a computation are shown in Figure 20. The difference caused by the two assumed model atmospheres is dramatic. For a given apoapsis, the required periapsis altitude may be reduced if an orbit lifetime less than the 50 year lifetime shown is permitted; however, the reduction is not as much as one might hope. As an example, for an apoapsis altitude of 50,000 km and a 20 year lifetime, the Model II periapsis altitude may be lowered from 3800 km to 3100 km. Clearly, it is essential to define a specific upper atmospheric density profile and an orbiter sterilization policy which are acceptable to the scientific community, both from the point of view of those wishing to make meaningful scientific observations as well as those desiring non-contamination of the planet.

Because of the relatively large ballistic coefficient of the orbiter as well as the large Sun-orbiter distance, solar pressure effects are negligible for the currently desired orbiter lifetime span of tens of years. Solar gravitational effects, however, are not negligible. Depending upon the orientation of the orbit plane with respect to the Sun, solar gravitational effects (often referred to as third-body effects) can cause the periapsis altitude to vary by as much as a few hundred kilometers if the orbit plane remains in certain undesirable orientations. Fortunately, the oblateness of Mars can be put to good advantage by causing sufficient continuous change in the orbit plane orientation such that solar gravitational effects will have a much lower chance of being accumulative. The result is that periapsis altitude alternately rises and falls over fairly short periods of time (on the order of one hundred days or so).

Summarizing, it appears that oblateness and solar gravitation and radiation effects are, by themselves, not too serious, unless they cause the periapsis altitude to diminish to regions where the atmospheric drag effect is significant. This latter effect depends upon the assumptions employed in defining a high-altitude density profile and, with the current large uncertainties involved, it is very important that the formulation of a standard high-altitude density model receive more attention. Finally, if one attempts to select a nominal periapsis altitude which is based upon (a) requiring a long orbit lifetime with a conservative atmospheric model, and (b) adding on an additional margin to accommodate solar gravitational effects as well as guidance and orbit determination errors, it is not at all unlikely that a nominal periapsis altitude as high as several thousand kilometers might be indicated. An altitude requirement of this magnitude may reduce some of the attractive features of performing an orbiter mission.

Potentially Attractive Satellite Orbits

Someone has said that "a fool can pose questions quicker than a wise man can find answers." I hope that is not the situation I find myself in today. There <u>are</u>, in fact, acceptable satellite orbits which <u>do</u> satisfy the various mission constraints, at least if we are willing to accept modest compromise.

Consider Figure 21. Recalling the previous discussion of approach geometry, the following comments can be made concerning the selection of preferred aiming regions about Mars:

(a) Region A is forbidden as a result of the contamination constraint which requires an adequate periapsis altitude to assure some minimum orbit lifetime for an unsterile orbiter. The size of Region A cannot be shown exactly due to the large uncertainties in the high-altitude atmospheric densities; however, the general size shown indicates that acceptable periapsis altitudes could be as large as one planet radius from the surface of Mars.

(b) Region B is underirable because orbit periapsis occurs on the dark side of Mars.
(c) Region C is sufficiently near the ecliptic to cause solar occultation at some time during each orbit.
(d) Region D results in the location of periapsis occurring at too extreme a northerly or southerly latitude, and the previously mentioned planetary coverage conditions are not met satisfactorily.
(e) We are thus left with regions E and F. Referring specifically to the Type I situation shown in Figure 21, region F is preferred over region E for the following reasons:

    (1) Region E does not cover the interesting latitude areas just south of the Mars equator; also, Canopus comes very near the Mars limb.
    (2) Region E does result, however, in a better Earth-occultation situation.
    (3) Region F results in excellent coverage of the southern latitudes as well as no Canopus loss due to the near Mars Limb.

Region F would therefore be recommended as the nominal Type I aiming region for this opportunity. Figure 22 shows the resulting latitude and lighting coverage afforded by a typical orbit in this region. Because the orbit selected has a period of some 42 hr compared to Mars period of 24 hr, the relative motion causes the sub-orbiter track to move from right to left for a direct (posigrade) orbit.

A three-dimensional view of such an orbit on the first day after encounter is shown in Figure 23. The interesting latitude and acceptable lighting area is outlined near the morning terminator. Another acceptable region exists near the evening terminator, but is out of sight of this view. The view shown is along the Earth-Mars line, and it can be seen that the Earth is occulted for a short time during the orbit. This occultation will end after the first week or so.

The final two figures show models of typical orbiter spacecraft configurations. The spacecraft in Figure 24 is shown in its "folded" position as it would appear during launch. The design is based upon a storable bi-propellant liquid propulsion system. Power is derived from extendable solar panels; a body-fixed high-gain antenna is employed. Figure 25 depicts a configuration using a solid retro propulsion system. The spacecraft solar panels are extended as they would be in their orbiting operation. As can be seen these panels, various other appendages, and the spacecraft itself will present field-of-view problems to the planetary scan instruments - but that is the subject of another lecture!

SUMMARY

In considering the design of planetary orbiting missions, particular attention must be given to selection of the orbit itself. Many factors must be carefully considered in order to arrive at an acceptable orbit profile. If the orbiter is to be unsterilized, one of the most important factors is the selection of a periapsis altitude which is sufficiently high to assure an acceptable minimum orbit lifetime. It is extremely important to define a standard high-altitude density model and an orbiter sterilization policy which are consistent with the mission objectives. Potentially attractive orbit profiles can be found which satisfy most of the desired mission constraints. Increasing periapsis altitude provides long orbit lifetime, improves the Canopus geometry, and delays or avoids Sun occultation; however, the resolution of Mars-oriented experiments will decrease, and the chances of performing an Earth occultation experiment diminishes. Finally, southerly direct orbit profiles appear attractive (in 1969) and result in good coverage of the preferred southerly Mars latitudes.

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2. Planetary and Space Science, Volume 9 July 1962, "Seasonal Evolution of the Fine Structure of the Dark Areas of Mars," J. H. Focas, pp. 371-381.

3. "NASA Engineering Models of the Mars Atmosphere for Entry Vehicle Design," NASA Technical Note TN-2525, November 1964.

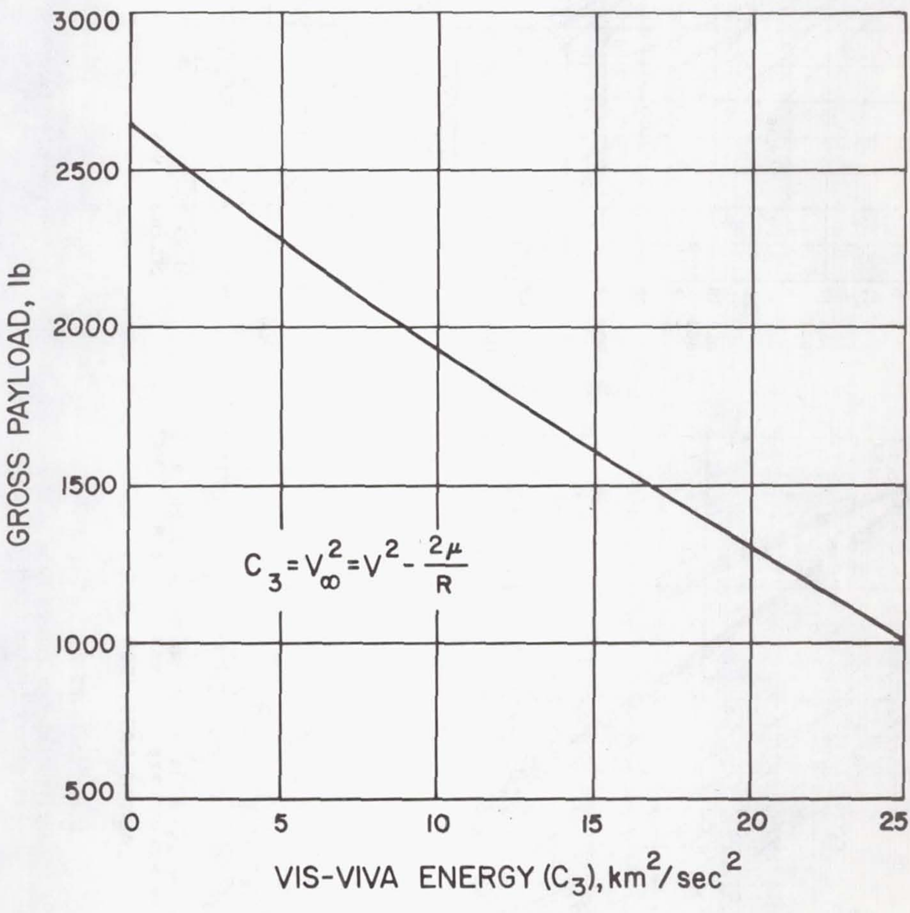

FIGURE 1

Typical Launching Vehicle Capability

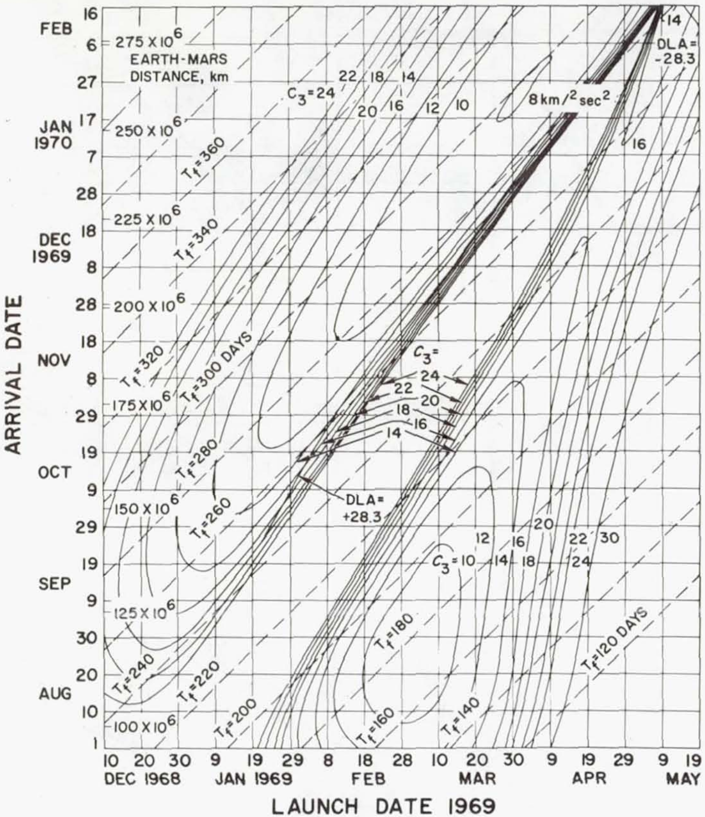

FIGURE 2

Arrival Date vs Launch Date
Trajectory Design Chart

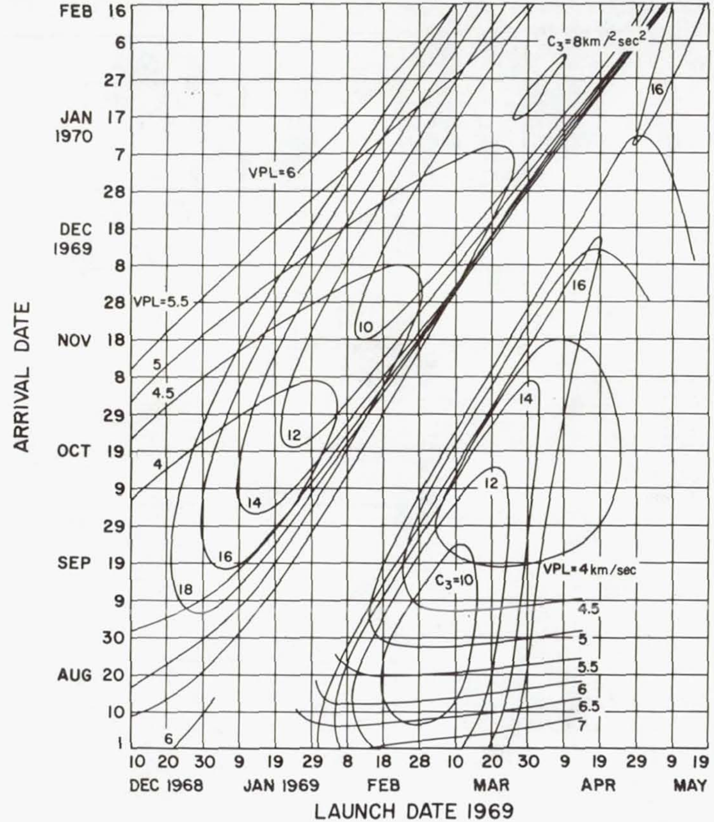

FIGURE 3

Hyperbolic Excess Velocity Relative To Mars

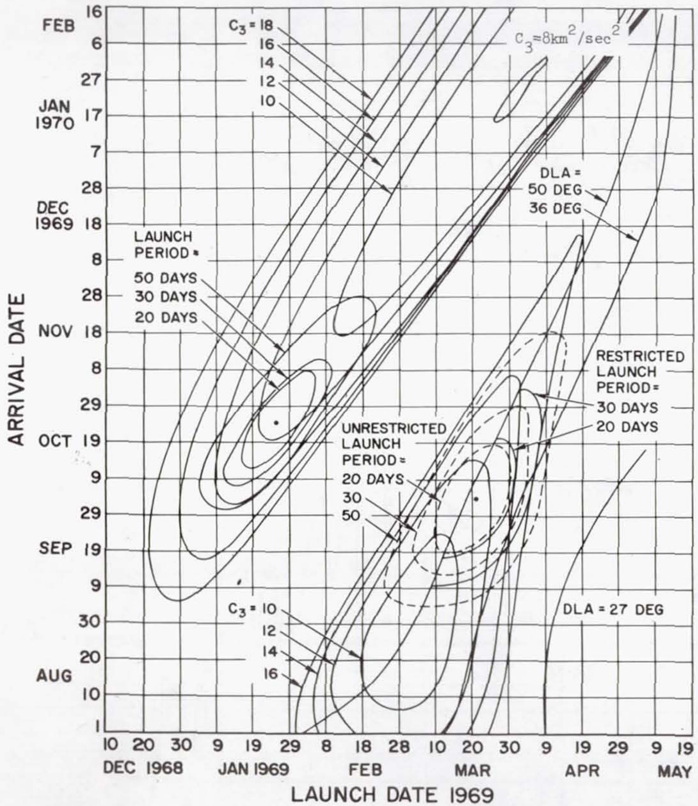

FIGURE 4

Curves of Constant Orbiter Mass For Various Firing Periods (Variable Propellant Loading)

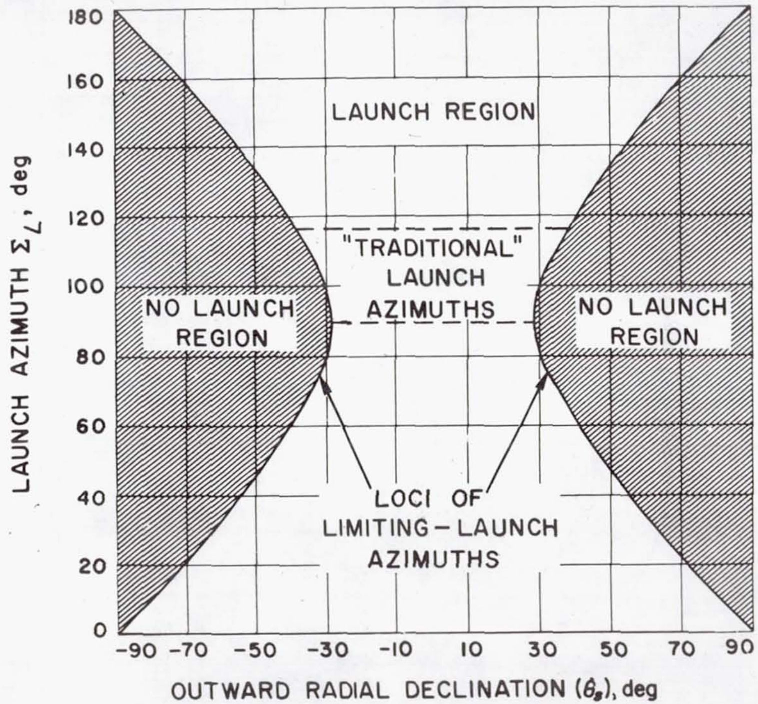

FIGURE 5

Limiting Launch Azimuth vs Outward Radial Declination For Cape Kennedy

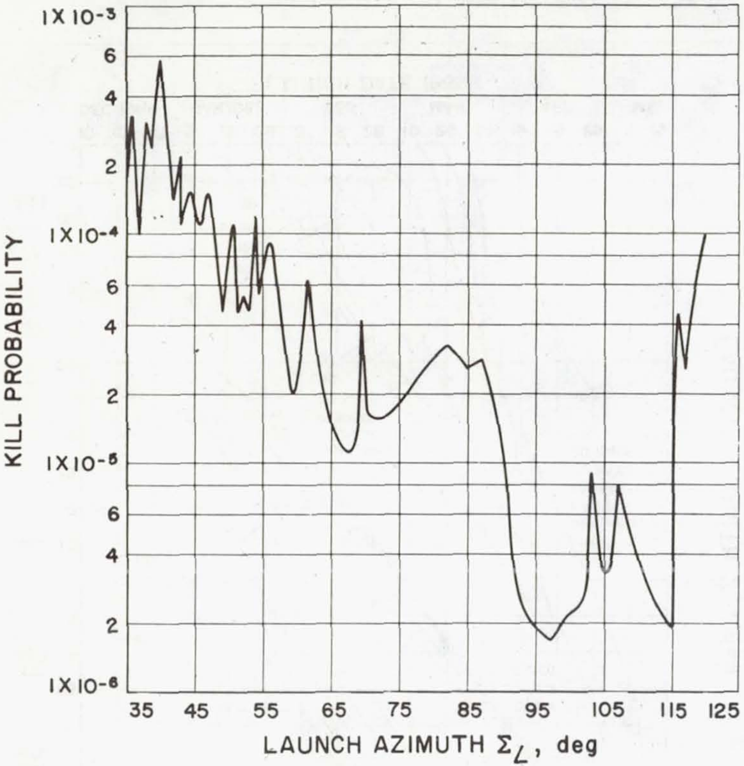

FIGURE 6

Kill Probability vs Launch Azimuth

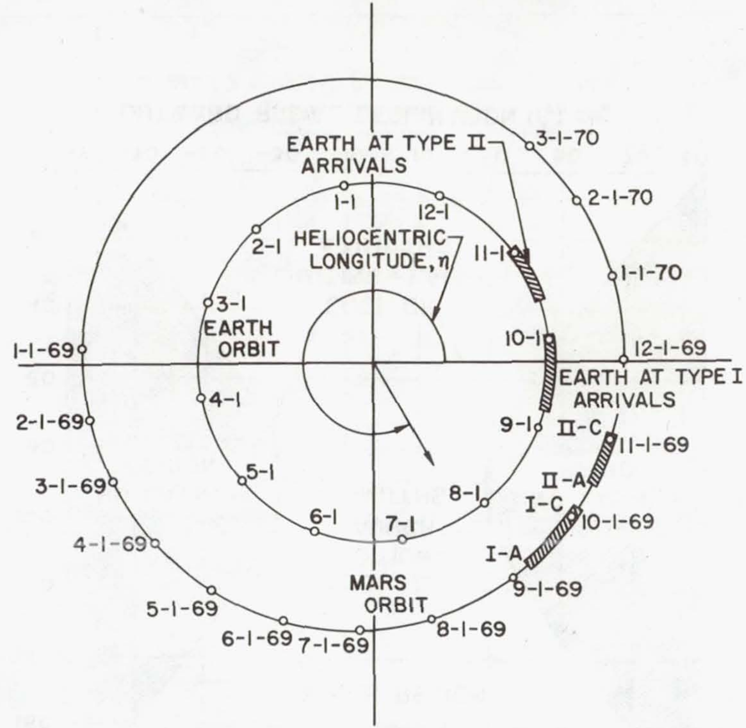

FIGURE 7

Heliocentric Arrival Geometry

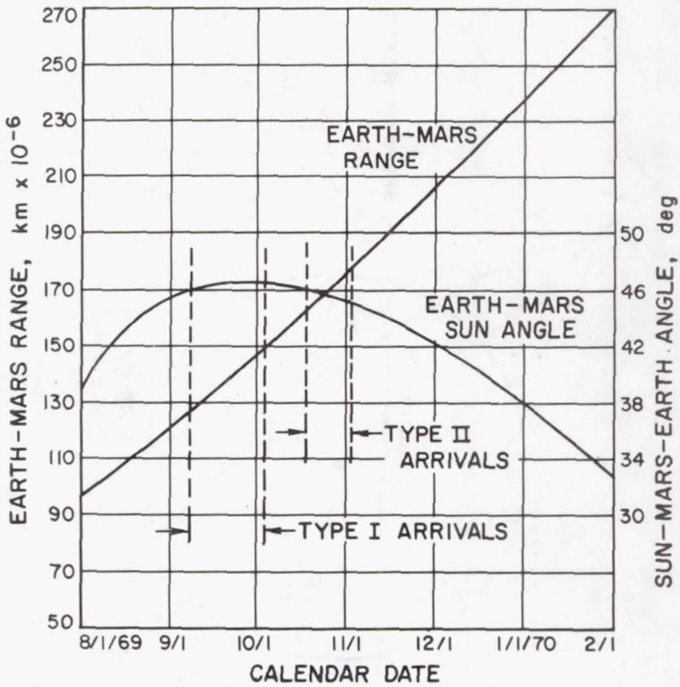

FIGURE 8

Earth-Mars Range, And Earth-Mars-Sun Angle vs Calendar Date

LAUNCH DATE (1969)	ARRIVAL DATE (1969)	FLIGHT TIME (days)	C_3 (km^2/sec^2)	VPL (km/sec)	CD_E (10^6 km) / CD_{E+180}	SD_E (10^6 km) / SD_{E+180}	η_{MARS} (deg)	TYPE
1-9	10-14	278	14.0	3.74	159 / 340	207 / 231	330	II-A
1-24	10-23	272	12.0	3.83	167 / 348	207 / 232	337	II-B
2-7	11-2	268	11.7	4.05	176 / 357	207 / 233	345	II-C
3-7	9-7	184	9.0	4.48	126 / 300	209 / 223	307	I-A
3-20	9-24	188	11.0	3.85	141 / 320	207 / 225	319	I-B
4-4	10-3	182	15.9	3.73	149 / 330	207 / 228	325	I-C

FIGURE 9

Summary Trajectory Characteristics

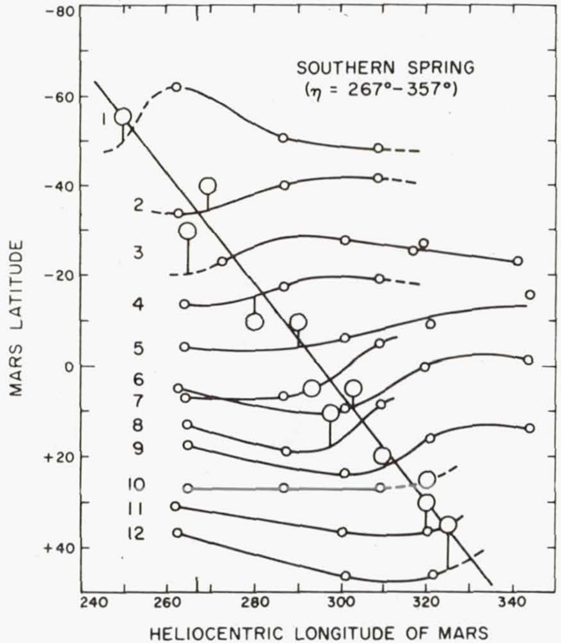

FIGURE 10

Seasonal Variations of Intensity in The Dark Areas of The Surface of Mars at Different Latitudes During The Southern Spring in 1939, After G. De Vaucouleurs (Peridlier Obs., 1945)

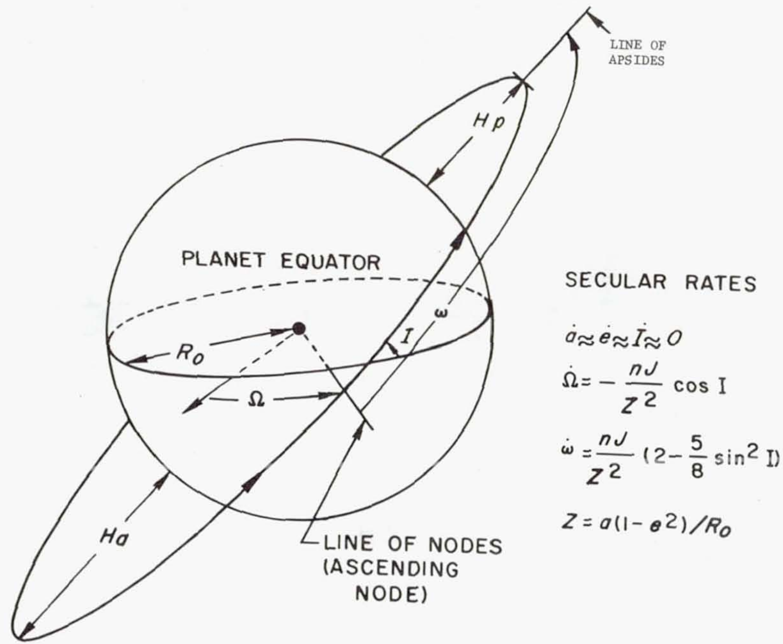

FIGURE 11

Fundamental Orbit Parameters

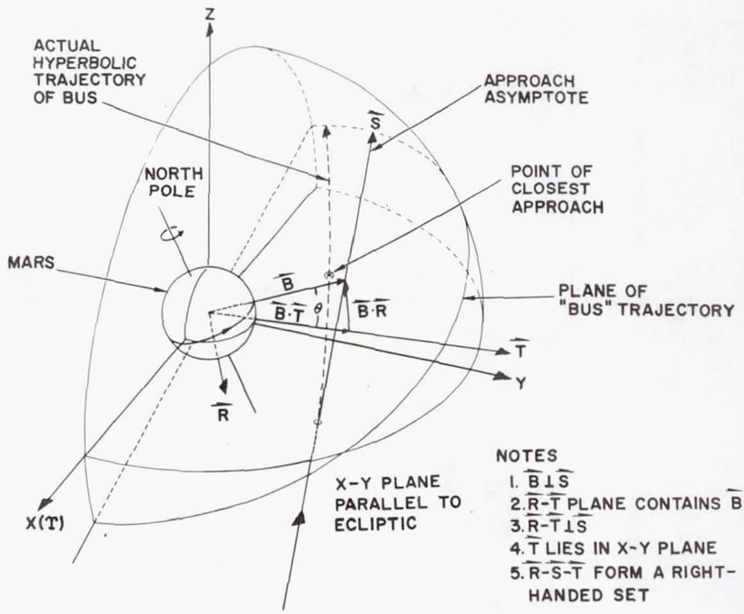

FIGURE 12

Description Of Miss-Distance Coordinates

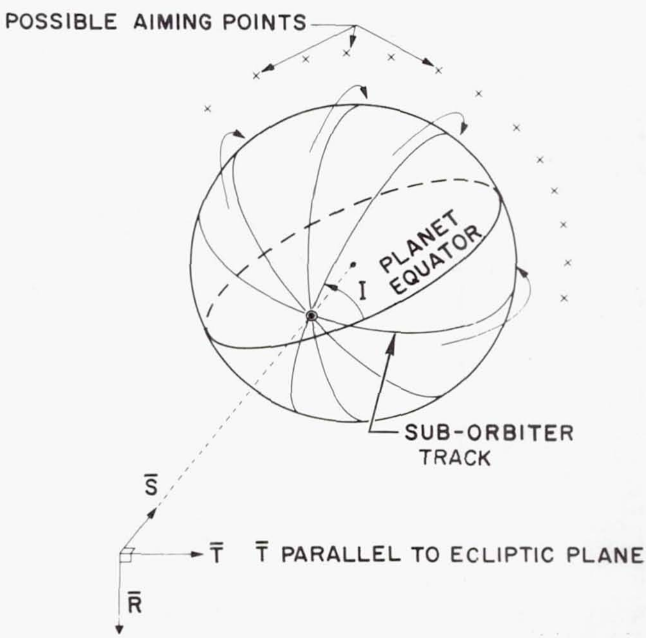

FIGURE 13

Rotation Of Possible Orbit Planes About Approach Asymptote

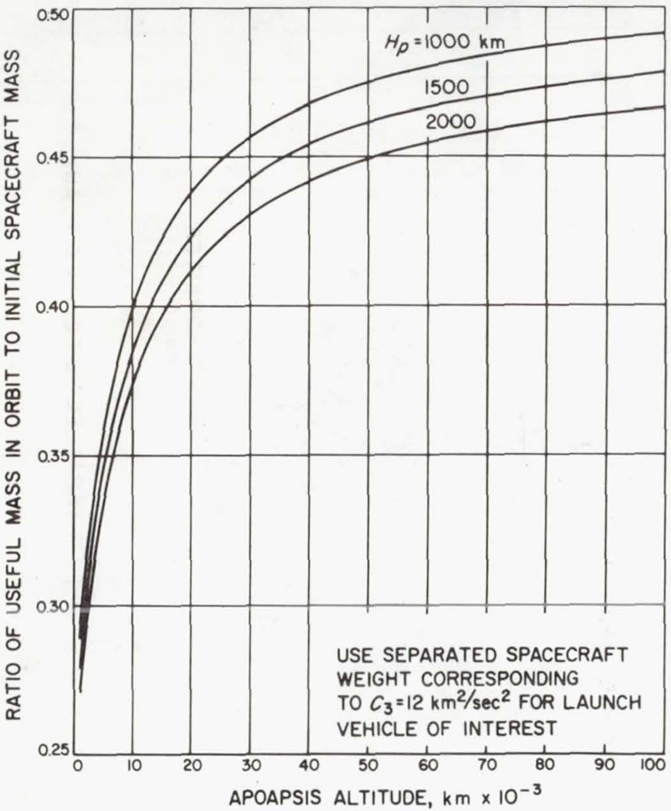

FIGURE 14

Orbiter Mass Ratio For Optimum Type II Trajectories With Variable Loading And 30-Day Launch Period

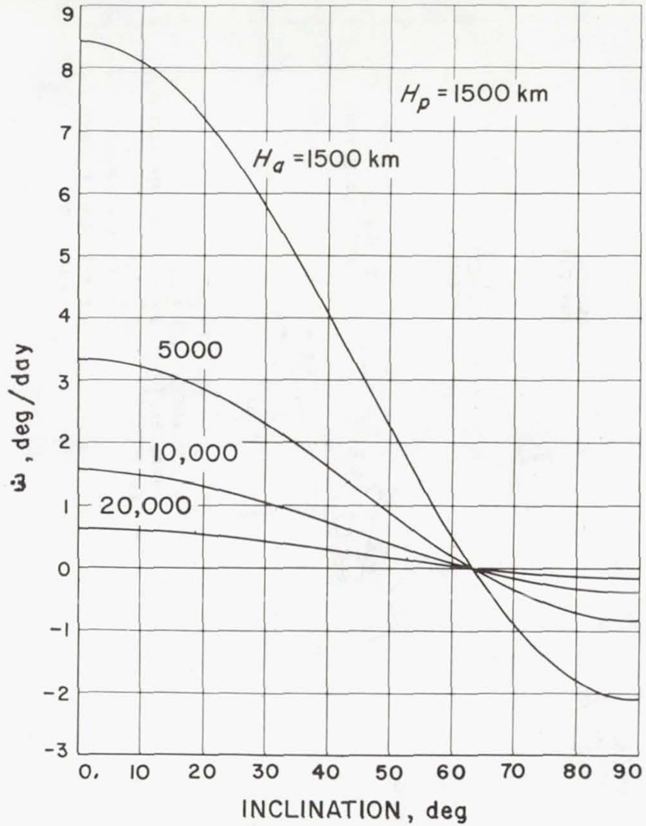

FIGURE 15

Apsidal Precession Rate vs Orbit Inclination To Mars Equator

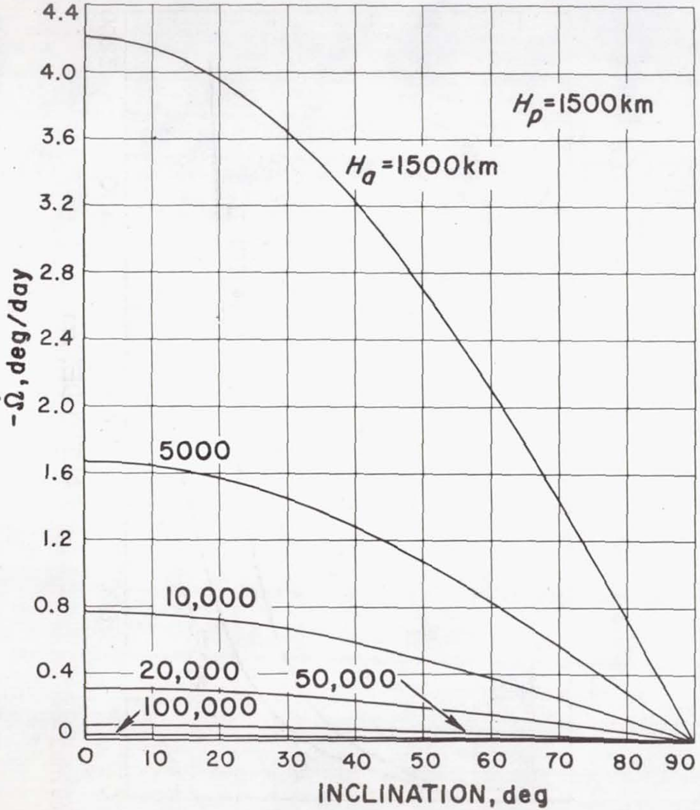

FIGURE 16

Nodal Regression Rate vs Orbit Inclination To Mars Equator

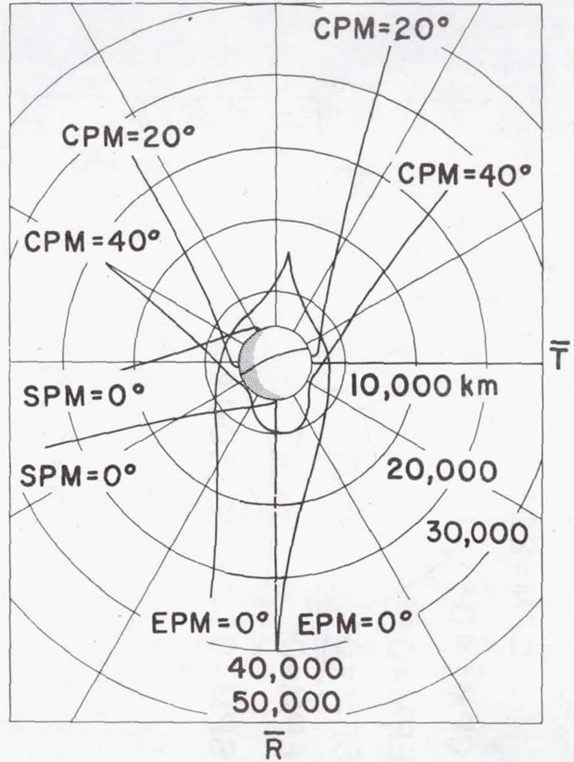

FIGURE 17

Approach Aiming Diagram For Transit 1-A

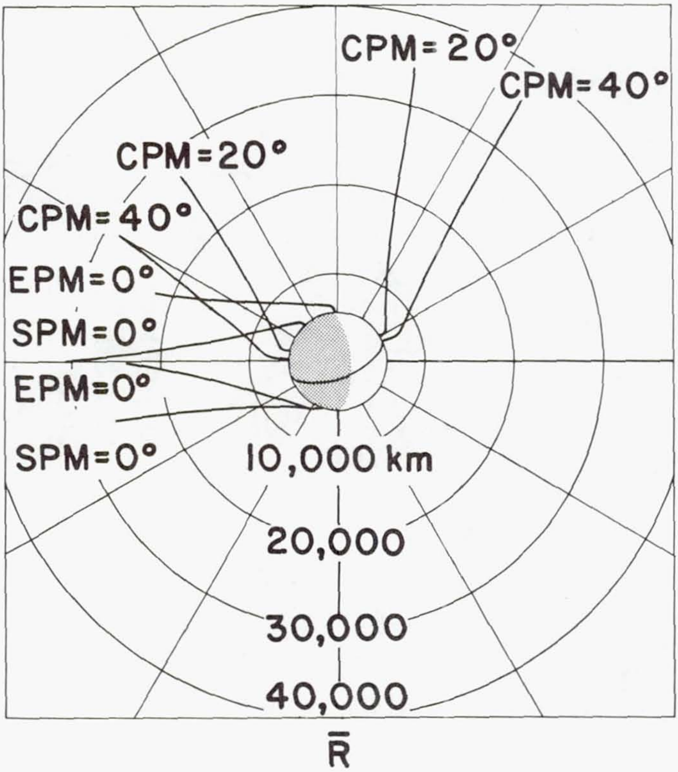

FIGURE 18

Approach Aiming Diagram For Transit II-B

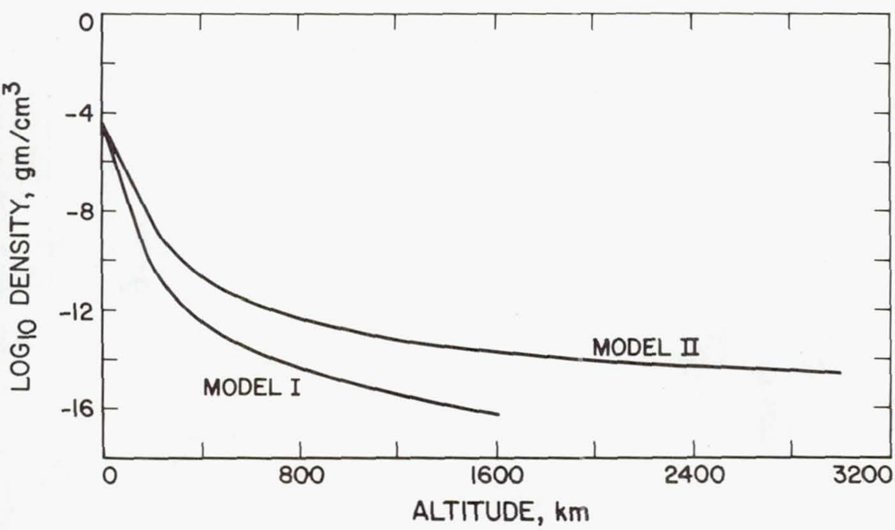

FIGURE 19

Log Density vs Altitude For JPL Model I And NASA "Maximum" Model II Atmospheres

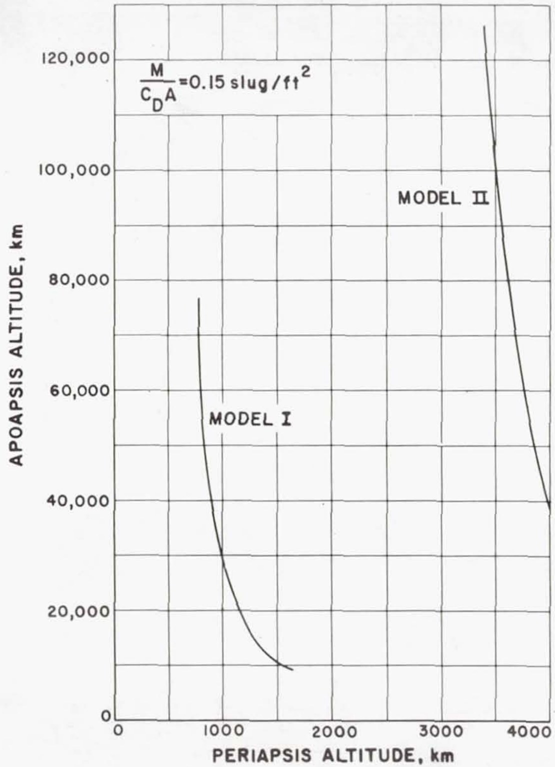

FIGURE 20

Apoapsis Altitide vs Periapsis Altitude To Yield 50-Year Lifetime

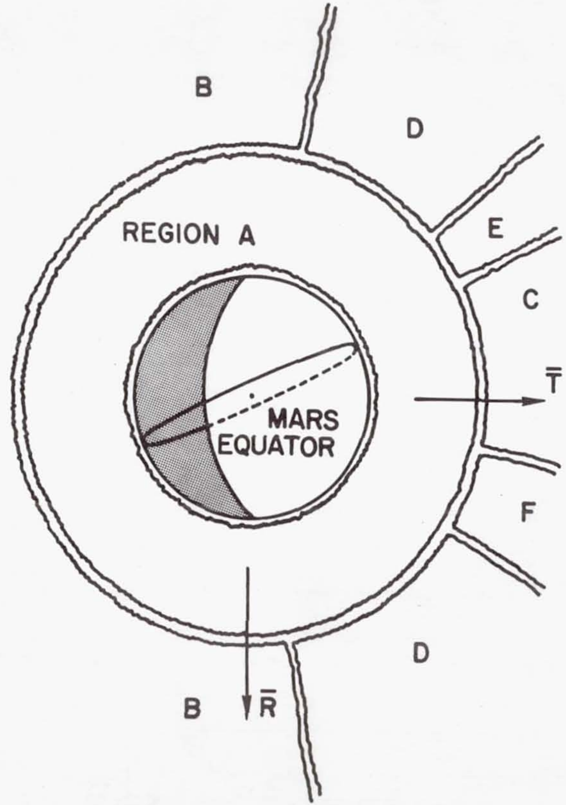

FIGURE 21

Type I Aiming Regions

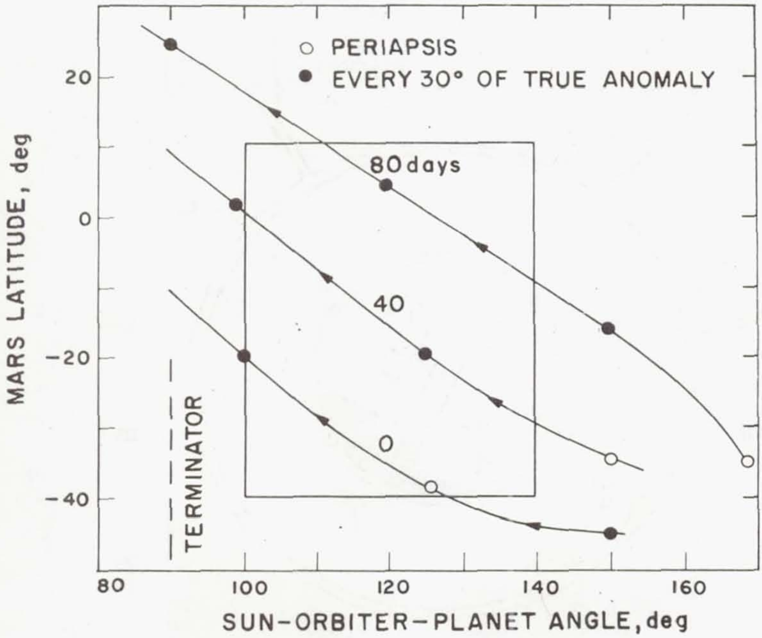

FIGURE 22

Latitude/Lighting Coverage For Type 1-B, 1500 By 50,000 km Orbit Profile Inclined 45 Degrees Below Mars Equator

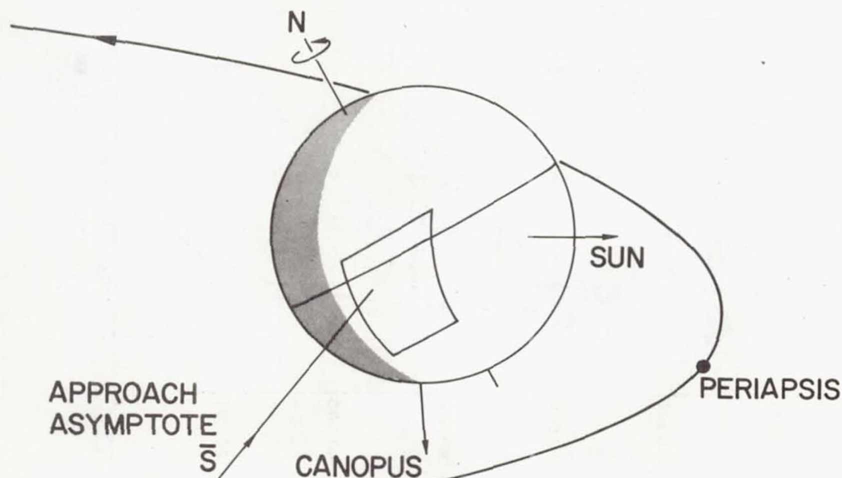

FIGURE 23

Type 1-B Nominal 4000 By 50,000 km Orbit Profile (View Along Earth-Mars Line)

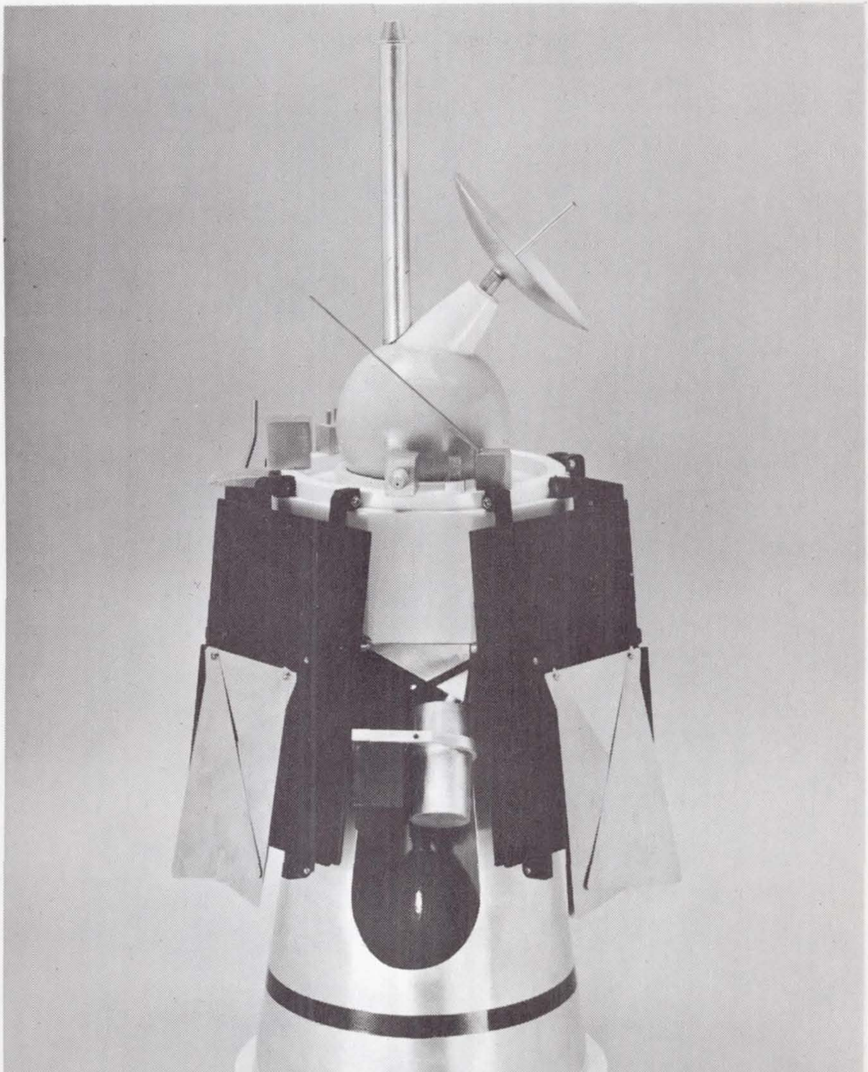

FIGURE 24

Model of Orbiter Spacecraft - Folded Position

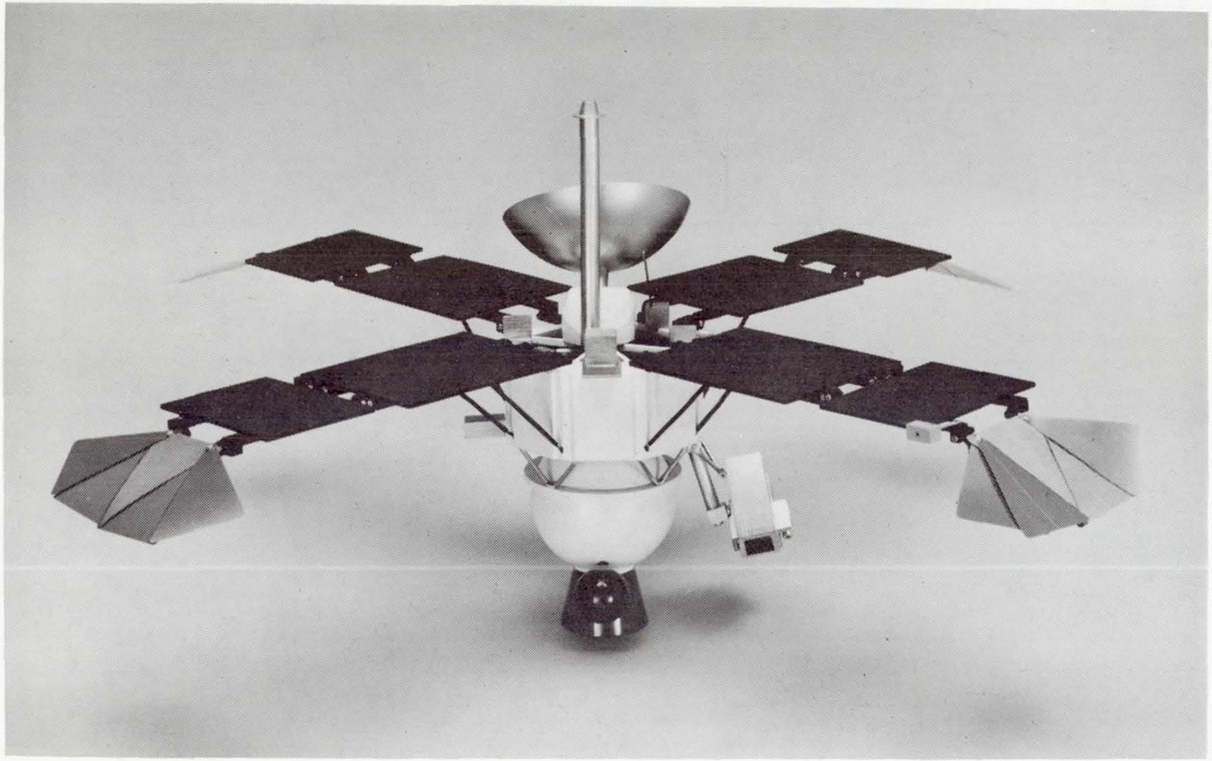

FIGURE 25

Model of Orbiter Spacecraft - Extended Position

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PROBE AND LANDER DESIGN PROBLEMS

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INTRODUCTION

The exploration of Earth's neighbor planet Mars, presents us with one of the greatest challenges of our time. It is of extreme interest, scientifically, to determine the character of the planetary atmosphere, the details of surface features, and the possibility of the existence of extraterrestrial life.

Detailed information can be obtained only by placing instruments into the atmosphere and on the surface of the planet and it is clear that spacecraft designed for atmospheric entry and surface landing will play an important role in the planetary exploration program.

In the last decade, a large research and development effort has been applied to the design of entry vehicles, for use in the Earth's atmosphere, and much of the technology produced can be applied directly to planetary exploration. There are some significant differences, however, between the needs of past and today's entry vehicles and what will be required for the future Martian program. These arise primarily because we face an atmosphere that differs both in pressure and composition.

This paper concerns research and development problems that arise in the design of unmanned planetary probe and lander capsules. Greatest attention will be given to Mars lander capsules, in view of the imminent needs of the planetary program.

PROBE AND LANDER CAPSULES

First, let us identify the major elements that comprise the overall spacecraft system: these can be identified as Flyby or Orbiter Spacecraft which may be placed near, but not in contact with, the planet or its atmosphere, and Atmospheric Probe and Lander Capsules which enter the atmosphere and land on the planet surface. The Flyby and Orbiter Spacecraft are capable of making measurements by observation over wide areas of the planet surface whereas the Atmospheric Probe and Lander Capsule can provide detailed information at specific locations. Clearly these major elements play complementary roles in the planetary exploration program.

Focusing now on the Probe and Lander Capsules (Figure 1), let us mention some of the measurements that are of interest from scientific and engineering points of view. First, the characteristics of the atmosphere itself are of extreme interest and bear on questions relating to planetary evolution, the nature of possible life forms and the understanding of energy balance in planetary atmospheres. From an engineering standpoint also, the atmosphere is important since future vehicles used in expeditions to the surface must enter the atmosphere and must be designed to withstand the associated forces and heating.

Surface measurements of interest include physical and chemical properties of the surface material, meteorological conditions, the presence of dust storms, diurnal and seasonal variations, possible biological growth or the evidence of past forms of life. Measurements of radioactivity and siesmic disturbances are also of interest since they help reveal the character of the interior.

The exploration of a planet whose atmospheric and surface environment is not well defined poses the problem of deciding what the first vehicle should be. An entry vehicle will be necessary for placing any measuring equipment into the lower

atmosphere and on the surface, but at the same time the atmospheric measurements are to some extent required for proper design of the entry vehicle. The first step must clearly be one that we can take with some confidence of success, and should yield information of scientific value - for example, composition of the lower atmosphere, which may shed light on extraterrestrial life, and other information contributing to design of future experiments. Moreover, the design concepts for the first mission vehicle should have growth potential - support subsequent, more elaborate vehicles.

Design of the entry vehicle must evolve from a number of inputs. Foremost among these should be a definition of the purpose of the mission, including the measurements to be made, the instruments to be used, and the manner in which data will be communicated. Then it will be possible to establish the internal payload, the required data-acquisition period, and permissible impact decelerations for a lander. Existing information on the planetary atmosphere - best estimates of atmospheric pressure, scale height, composition, etc. - together with the expected entry conditions, particularly entry velocity, will determine within reasonable bounds the problems associated with entry loads, entry heating, communications blackout, and terminal impact. These factors place further constraints on the configuration.

As a point of departure, consider a spherical atmospheric probe, suggested by Seiff[1], and more recently elaborated by Buef[2], for an early mission, say 1969. This spherical probe would measure atmospheric properties through its own motion during entry. The deceleration, which is independent of the attitude for a sphere, would be measured and transmitted to the parent flyby spacecraft for relay to Earth, data communication taking place over a period of about 30 sec. after the probe emerged from blackout. Velocity during entry would be derived by integrating the deceleration history, and the altitude by further integrating with respect to time. From this information the variation of density and pressure can be determined from the drag equation. Additionally, some information regarding the composition can be determined by observation and analysis of the spectrum of radiation emitted by the shock layer that surrounds the sphere during entry.

Such an atmospheric probe appears feasible now if it can be placed on the appropriate flight path to ensure a fairly steep entry angle; but, since the atmospheric properties are not measured directly, the data would not be very accurate.

It is clear that more extensive measurements will be required, both in the atmosphere and on the planet's surface, with correspondingly longer periods for data communication. At the 1971 launch opportunity, Saturn-Centaur should be able to deliver a Probe/Lander to determine, in detail, conditions in the lower atmosphere. Preliminary information on surface conditions would be obtained while the lander remained in view of the flyby spacecraft. Such a Probe/Lander would be a very lightly loaded, high-drag vehicle, and would require a terminal parachute to decelerate sufficiently in the tenuous Mars atmosphere. Data from such a mission would support the formulation of experiments and design of sensors and other instruments for future missions, possibly an automated biological laboratory or an automated weather station. These advanced vehicle would soft-land retrorockets, operate for many months, possibly a year, and communicate directly with Earth or use a planetary orbiter as a relay link.

DESIGN CONSIDERATIONS

Perhaps the most significant factor to affect the design of probe and lander capsules is the atmospheric pressure at the planet surface - this factor in effect determines the extent to which the atmosphere can be relied upon to provide deceleration to the capsule. Figure 2 describes the history of a capsule during vertical entry into a planetary atmosphere, velocity being plotted against an atmospheric drag parameter (essentially a ballistic parameter introduced by Allen and Eggers).[3] This drag parameter can be interpreted in the following way. During the time in which the capsule decelerates from the entry velocity, V_e, to a velocity V, it encounters a mass, m_{atm}, of the atmosphere. The drag parameter is simply the ratio m_{atm}/m and can be written

$$\frac{m_{atm}}{m} = \frac{\int \rho C_D A dh}{m} = \frac{p}{mg/C_D A}$$

where ρ, p, and h are respectively the density, pressure, and altitude in the atmosphere, g is the planet gravitational acceleration, and $C_D A$ is the effective drag area of the capsule.

During atmospheric entry, the vehicle undergoes aerodynamic heating and sustains aerodynamic loads (these are most severe at m_{atm}/m equal to 1/3 and 1, respectively), continues to decelerate until it passes through sonic conditions, and finally, if there is sufficient atmosphere, reaches a terminal condition (m_{atm}/m greater than about 10 typically) in which the drag force is balanced by the gravitational force.

It is clear from the figure (Figure 2) that the vehicle velocity decreases rapidly as the ratio m_{atm}/m or $p/(mg/C_DA)$ becomes larger, and it follows that the value of m/C_DA required to decelerate the vehicle to terminal conditions varies directly as the atmospheric pressure, p. Now for Mars, the atmospheric pressure near the surface is presently thought to be as low as 10 millibars (compared with approximately 1000 millibars for Earth) and as a consequence terminal conditions are achieved only if m/C_DA is less than about 0.2 slug/sq ft.*

The design parameters which characterize the spherical probe are the ballistic parameter m/C_DA and the diameter D, and it is convenient to discuss the sphere in terms of these quantities. The constraints which determine the permissible range of these parameters depend on the entry velocity and angle and on the atmospheric properties. The expected entry velocity, dictated by the interplanetary trajectory, would be known fairly accurately, and for the present it is assumed that the entry path is vertical. If the atmospheric properties are assumed known, as they must be to arrive at a design, then all the factors which determine the constraints can be expressed in terms of m/C_DA and D.

Taking first the case of Mars, the most significant constraint results from the low atmospheric pressure at the surface. A sphere of conventional mass-to-area ratio - m/C_DA = 1 slug/sq ft - entering the Mars atmosphere (having a surface pressure of 10 millibars) at 25,000 fps would spend only a few seconds in it before impacting the surface at about 10,000 fps. Furthermore, under these conditions it would suffer communication blackout throughout most of its flight and would impact before emerging from blackout. Clearly, if the probe is to have sufficient time to gather and communicate data, it must have a value of m/C_DA significantly less than 1 slug/sq ft. As the value of m/C_DA is reduced, the available data-communication time is increased; and for m/C_DA = 0.25 slug/sq ft the period between emergence from blackout and impact is about 30 sec. It seems unlikely that a significant amount of information can be obtained in less time than this.

To obtain a longer data-collection period, the value of m/C_DA must be further reduced, and for a given drag area, C_DA, this implies a reduction in payload mass. Here, however, it should be remembered that the sphere must withstand both aerodynamic loads and aerodynamic heating, and a limit is soon approached in which the entire mass is assigned to the load-carrying structure and to thermal protection, leaving no mass available for payload.

The aerodynamic loads for vertical entry into the Martian atmosphere, at a typical entry velocity of 25,000 fps are expected to be about 200 Earth-g and the convective heating approximately the same as that for Earth entry. Additional heating associated with radiation from the gas layer surrounding the vehicle depends on the atmospheric composition. For N_2-CO_2 mixtures, likely candidates for the atmospheres of Mars and Venus, it has been shown that radiative heating is significant at entry speeds as low as 20,000 fps, whereas for Earth entry this form of heating is not significant even at 30,000 fps. For spheres of large diameter the structural weight increases more rapidly than the surface area, due to the increased thickness of the structural shell required to maintain stiffness and so avoid compressive buckling; and increased radiative heating places greater demands on the thermal-protection system - radiative heating per unit area increases directly as the diameter.

For spheres of small diameter, the structural weight becomes less significant; but the convective heating, which varies as $D^{-1/2}$, requires a greater thickness of ablation material, and the low m/C_DA sphere can do little more than provide itself with sufficient protection to survive entry heating. Figure 3 which plots m/C_DA vs. D on lagarithmic scales, indicates how the designer is literally boxed in by these constraints when he attempts to design a spherical entry vehicle. The lower boundary corresponds to an internal payload of 1/2 slug representing about the minimum mass required for a power supply and communication system. Along the upper boundary of the design box, where m/C_DA = 0.25, the minimum diameter is 8 ft, corresponding to a payload of 3.5 slugs, the maximum possible for any sphere within these constraints. Spheres of larger diameter would be structurally too heavy to carry 3.5 slugs and stay within the m/C_DA constraint. The Centaur shroud diameter,

*1 slug = 32.2 lbs.

shown in the chart, does not present a significant constraint, since the larger spheres would correspond to reduced payloads.

The Mars spherical probe, in short, appears to be feasible if the internal payload has sufficiently small mass and if the data-collection period can be of the order of 30 sec; but it offers little promise of being useful for the larger payloads envisioned in future missions.

Considering now the Probe/Lander Capsule,[4] depicted in Figure 1, the primary constraint arises from the need to achieve a sufficiently low velocity to allow deployment of a parachute. With a conventional parachute, the vehicle must decelerate to subsonic speeds before reaching the surface, and this requires that m/C_DA be less than 0.2 slug/sq ft. If the vehicle must collect appreciable atmospheric information before landing, it is desirable to reduce m/C_DA further, to a value of about 0.15 slug/sq ft.

The limitations of the sphere as a low-m/C_DA entry vehicle lead us to consider other shapes which for one reason or another are more efficient. For a given area, A, the allowable mass increases with the drag coefficient, and the mass of the internal payload increases further when the structural weight and the thermal-protection weight - both of which vary as the surface area - are minimized.

Ideally, then, the vehicle should have high drag and small exposed surface area and still enclose sufficient volume to contain the payload. A flat disc has the highest drag ($C_D \approx 1.8$) for given surface area but has no volume, whereas the sphere contains the maximum volume for given surface area but has insufficient drag. Evidently the ideal vehicle would combine a flat disc with a sphere of sufficient volume to contain the internal payload. In general, because the required mass-to-area ratio must be small, the sphere needed to contain the payload has relatively small diameter, compared with that of the disc. A combination of a disc and such a sphere, suitably rounded off to make it aerodynamically respectable, leads to either an Apollo-shaped vehicle or to a shallow blunted cone, as indicated in Figure 4. Such shapes have drag coefficients of the order of 1.5 and have low exposed surface areas, S, leading to low thermal-protection requirements. (Generally speaking, a small value of the parameter S/C_DA is desirable, and the shallow blunted cone has a value 0.8 compared with 2 for a sphere.) Three conceptual Capsules now under study are illustrated in Figure 5; the blunted cone tends to have better aerodynamic stability than the Apollo shape, especially if the internal payload mass can be placed at the bottom of the cone, and for this reason the trend for future unmanned Mars vehicles is likely to be in this direction.

Having reduced the total surface area in this way, further increases in payload can be realized only by reducing the structural and thermal-protection weights per unit area. The thermal-protection material must itself satisfy a number of requirements. For example, it must be capable of withstanding the long "cold-soak" experienced during the space flight; it must have high thermal performance during entry; and, if possible, it should allow communication after entry. Such materials (for example, elastomeric materials) are presently available, but the choice is extremely limited, and there is little possibility of reducing the thermal-protection weight by any appreciable amount.

Structural weight, especially for vehicles of large diameter, becomes the most significant part of the total vehicle weight for any structure under compressive loads. As Figure 3 shows, this is clearly so for a sphere, since increasing the diameter merely increases the structural weight and actually reduces internal payload. The "tension cone" illustrated in Figure 5 represents an attempt to reduce structural weight to a minimum by shaping the vehicle so that it is in tension when aerodynamically loaded, and is therefore not subject to compressive buckling. The advantages of this shape, however, are offset by the more complex aerodynamics associated with flow separation and shock unsteadiness.

Figure 6 shows the internal payload mass available, comprising the payload and retardation system, when Capsules such as those illustrated in Figure 5 are used. It also shows the ranges of m/C_DA requiring parachute and retrorocket systems to achieve a survivable landing. A Probe/Lander Capsule, 15 ft in diameter, designed to achieve subsonic speeds at 15,000 ft in a 10 mbar atmosphere (i.e., $m/C_DA = .15$ slugs/sq ft) would be limited to an internal payload of 12 slugs (approximately 400 lbs.)

More advanced vehicles, such as an automated biological laboratory or an automated weather station, may be appreciably heavier than an early Probe/Lander Capsule; such increases in internal payload weight are allowed only if the surface pressure is found to be greater than 10 mbars, or if the entry angle is reduced

permitting the Capsule to traverse a longer flight path in the atmosphere and thereby sustain more aerodynamic retardation. Figure 7 illustrates the Capsule payload growth potential associated with these effects; a shallow entry angle (30°) or 20 mbar atmosphere permits internal payload growth up to 1,600 lbs whereas the combination of these factors permits growth up to 4,000 lbs.

If the atmospheric pressure is indeed found to be 10 mbars the restriction to aerodynamic retardation after direct entry from the interplanetary trajectory still places a severe constraint on payload mass, and it may be appropriate to use some propulsive retardation.

The gains to be realized by propulsive retardation after descent through the atmosphere are not very great however since the additional mass required for propellant fuel causes the atmospheric retardation to be less effective; typically, increases in useful payload mass associated with such propulsive retardation are less than 30%.

An alternative approach involves the use of a supersonic decelerator prior to the deployment of a conventional parachute; here again, however, in order to be effective such decelerators must be large in area or deployable at very high Mach numbers and in either case tend to add additional mass to the system. Realistically, potential gains in useful payload mass are at most 25%.

The discussion thus far has indicated that it is feasible to decelerate to subsonic speeds approximately 400 lbs of internal payload even if the atmospheric pressure at the surface is only 10 mbars. This internal payload consists of a parachute, impact attenuation system and a scientific data acquisition system, i.e., instrumentation, power and communication system. The relative weight fractions of these elements depends on the maximum allowed values of the impact velocity and impact acceleration; for an impact velocity of 100 ft/sec vertically the parachute weight is approximately 10%-15% of the suspended payload and for decelerations of the order of 1,000 Earth g's the mass of a passive impact attenuation system, i.e., crush-up material is typically 30-40% of the impacting mass.

The resulting residual weight of the scientific data acquisition is therefore only about 200 lbs to 250 lbs depending primarily on the kind of crush-up material and the shape of the impacting package.

Finally when the weight of communication and power subsystems are accounted for together with the structure required for the lander package, there remains approximately 50 lbs available for scientific and engineering instruments.

TECHNOLOGY REQUIREMENTS

The capsule system must survive and operate over a wider range of environmental conditions than any previously met in the space exploration program. Even before launch it is subjected to heat sterilization of 135°C for 24 hours; during launch it encounters severe vibratory loads, and during its eight month spaceflight it may suffer micrometeoroid impact, solar radiation and thermal extremes. Upon arrival at the planet it faces atmospheric entry with its associated severe heating and aerodynamic forces, followed by impact on an unknown surface at high speed. Only then can the scientific equipment begin to make measurements and communicate the results back to Earth over distances in excess of 100 million miles.

Clearly many technologies must meet and be brought to bear on the design of the Capsule and in many respects the design requiremtnts call for unique technical approaches. The successful development of entry vehicles for the planetary exploration program depends to a large extent on the strength of the supporting research and technology program. Instrumentation for the first missions has, to a large degree, already been developed, and in some cases flight-tested. But the question of the performance of these instruments after sterilization - at present a requirement for all planetary vehicles - is still unanswered. The state of the art for communications from an entry vehicle allows data transmission before and after "blackout" with little difficulty. In recent years, considerable experience has been gained in this area in connection with Earth-entry flight programs, and the techniques developed there can be used with little modification. The aerodynamics of the spherical probe are, of course well known; but more-complex shapes require further research on the aerodynamics (drag, stability, convective and radiative heating through a large range of angle-of-attack), on the stability of lightweight structures, and on aeroelastic effects that may be associated with vehicle oscillation during entry.

Similarly, further research and development is required to produce a sterilizable parachute deployable at the low dynamic pressure levels anticipated in the Martian atmosphere, although considerable information exists on parachute performance under more conventional conditions.

Again, in the area of impact attenuation a great amount of data exists on crushable materials and on pneumatic energy absorbing devices, but there is little experience in the operation of scientific instrumentation and communication equipment deployed from within such devices after impact.

There are many questions not yet answered but there is little doubt that a vigorous technology program will yield the necessary design solutions and will permit a landing on the surface of Mars in the early 1970's.

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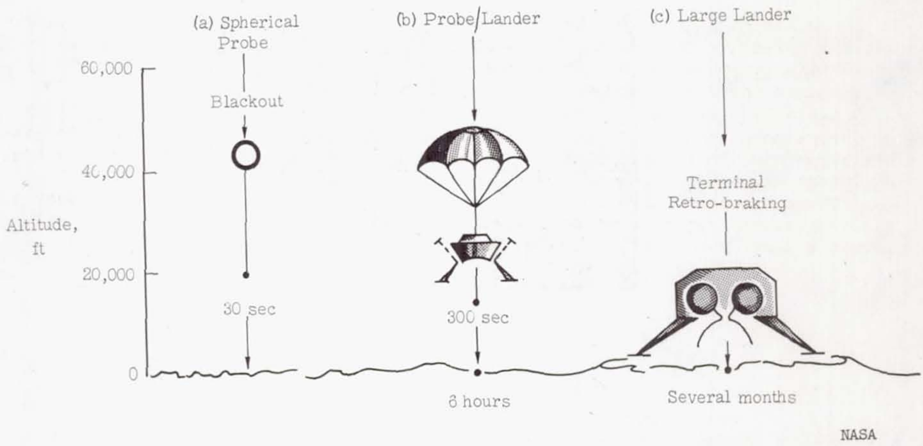


FIGURE 1

Mars probe and lander capsules.

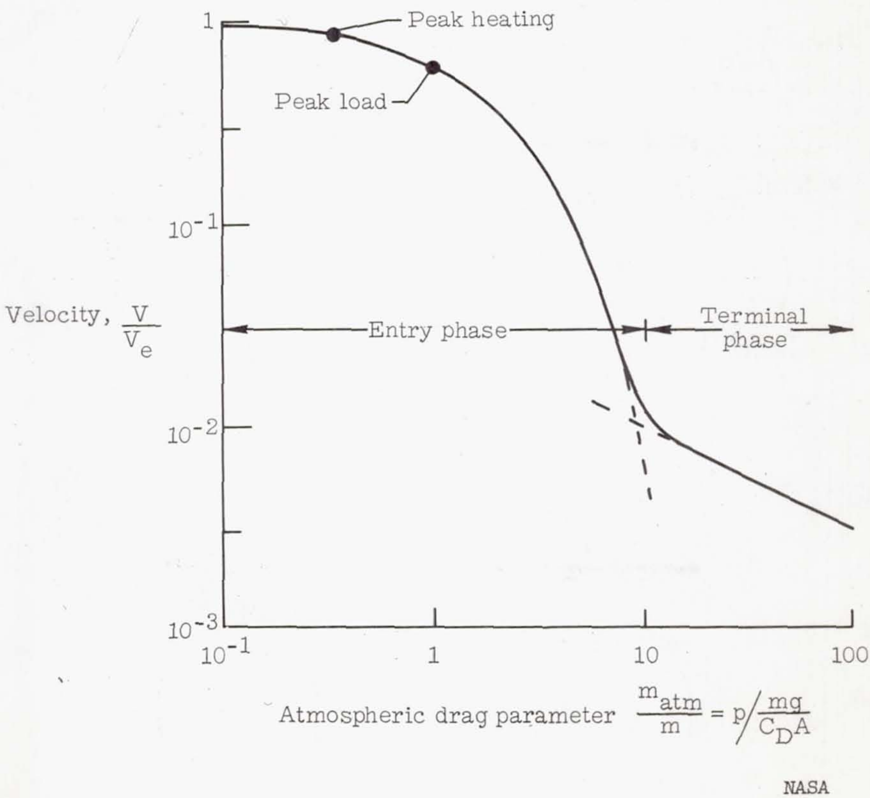


FIGURE 2

Atmospheric drag parameter.

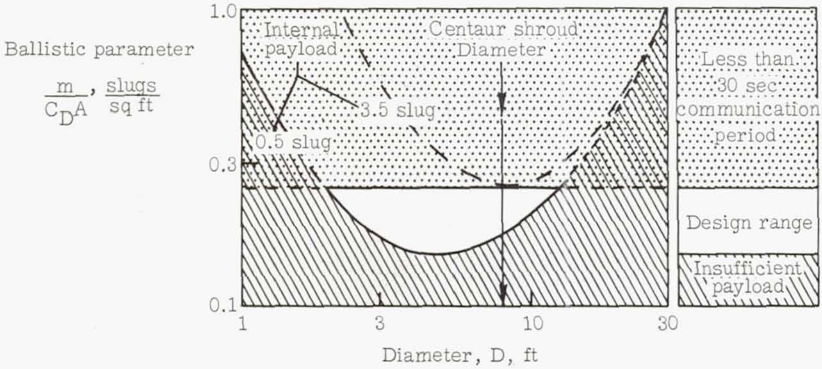
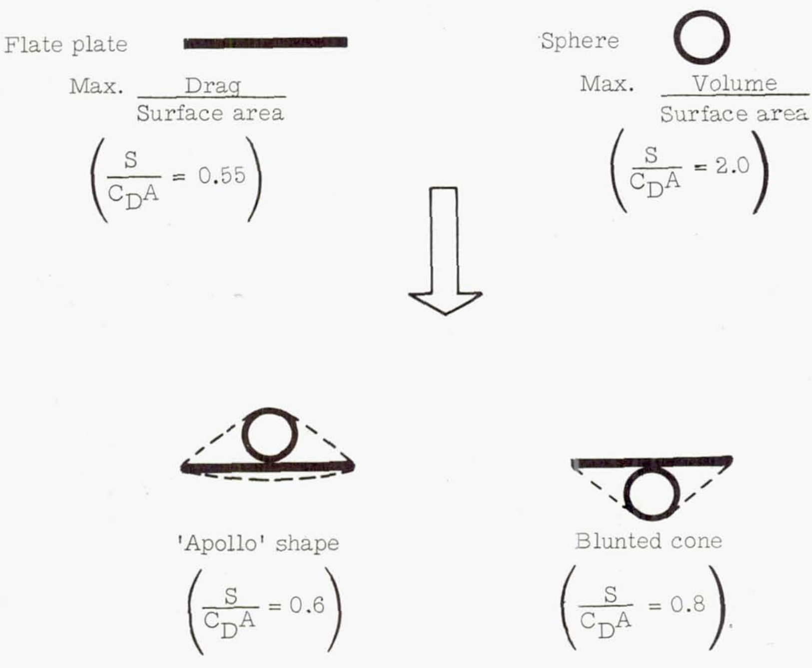


FIGURE 3

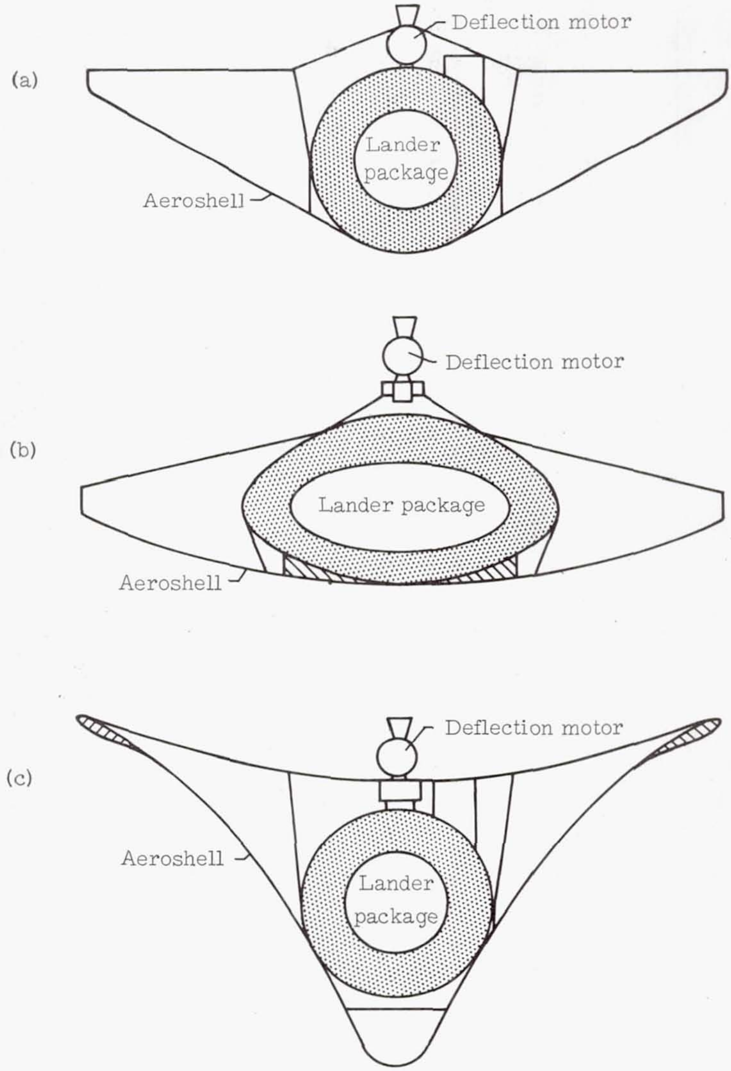
Constraints on spherical probe design. (Atmospheric pressure, 10 mbars; vertical entry at 25,000 ft/sec.)



NASA

FIGURE 4

High-drag capsule shapes.



NASA

FIGURE 5

Probe/lander capsule concepts.

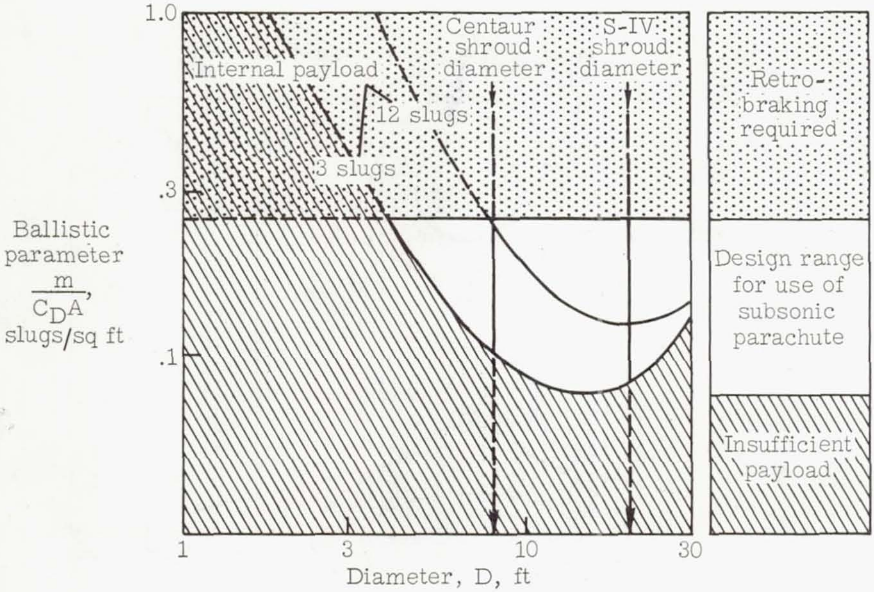


FIGURE 6

Landed payload weight for high-drag capsules. (Atmospheric pressure 10 mbars; vertical entry at 25,000 ft/sec.)

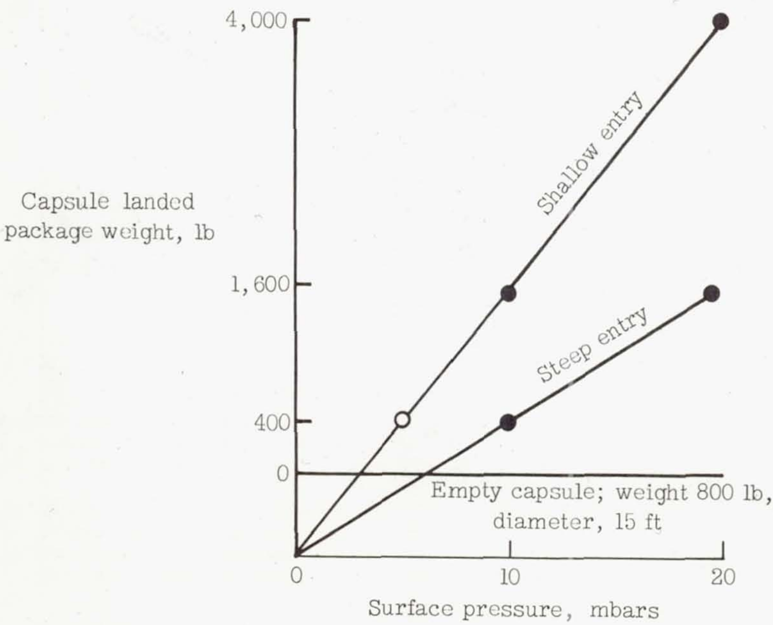


FIGURE 7

NASA

Landed weight growth potential, 15-foot-diameter capsule.

SPACE - QUESTIONS AND ANSWERS

By

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INTRODUCTION

I wish I had kept a log book of all the tough questions I have been asked about the space program during the past seven years. They would provide an interesting chronicle of the onrushing tide of events. The original answers might seem alternately wise and foolish today.

This thought has led me to depart from the single theme format of a typical luncheon talk. Instead, I'm going to answer a series of oft-repeated questions from laymen and experts alike.

1. Why explore space?
2. Is there really a space race?
3. Who has made the greatest strides in space?
4. Can we afford a civilian space program?
5. Why have both a civilian and military space program?
6. Will the unmanned program be swallowed up by the manned program?
7. What is the single most difficult problem in carrying out the space program?
8. Are incentive contracts just another gimmick?
9. What is the role of nuclear energy in the space program?
10. Why confine ourselves to Mars and Venus when the whole solar system is accessible?
11. When will we have manned planetary exploration?
12. Have we compromised our scientific integrity in "selling" our programs to Congress?
13. Do many scientists really oppose the space program?

WHY EXPLORE SPACE?

This perennial question has no single simple answer. It is the multiplicity of answers which is so overwhelming. We explore space in the name of science, in the name of security, and in the name of adventure. By so doing, we are building brain, and muscle, and spirit.

Only by tackling the most difficult and challenging proglems of our day can this country maintain its leadership in science and technology. Once earned, national greatness must be maintained by continuing accomplishment and service. The pages of history will one day record space exploration as one of our most noble contributions to civilization.

IS THERE REALLY A SPACE RACE?

With the establishment of NASA in 1958, T. K. Glennan said, in effect: Our job is to see that this country is second to none in space. Let's do it! There is no place for tired men in the space program.

Few who were in that initial gathering will forget Dr. Glennan's admonishment. We did set out to excel in space. We're still trying, and I think succeeding. Dr. Glennan was wrong on only one point. There is plenty of room for tired men in the program -- so long as it doesn't affect their work.

WHO HAS MADE THE GREATEST STRIDES IN SPACE?

Here I cast my vote with the home team. In the important area of the practical application of satellites, there has really been no contest. This country was quick to recognize the potential of satellite meteorology, communications, and navigation. Sufficient R&D has already been accomplished with TIROS, Relay, SYNCOM, and the Navy's navigational satellite system, so that operational systems in all three areas will be flying this year; the Navy with its system, the Weather Bureau with TOS, and the ComSat Corporation with Earlybird. The Air Force communications satellite system will not be far behind.

The NASA Nimbus and the Applications Technology Satellite (ATS) are laying the groundwork for even more effective systems in the future. We have just begun to realize the ultimate utility of space flight.

In the space sciences, our Explorers, monitors, and observatories have shown a breadth and depth unmatched by the Soviets. Admittedly, we don't know everything they are doing, but, based on what we know and the results of their work in the scientific journals, we are learning more about space than the Russians. Recent highlights of our scientific satellite program have been the Interplanetary Monitoring Platform (IMP), the Orbiting Solar Observatory (OSO), and the Orbiting Geophysical Observatory (OGO).

In the lunar, planetary, and interplanetary area of space sciences, we also lead in actual accomplishment. Pioneer V was the first truly successful monitor of the interplanetary medium. Mariner II to Venus was the first successful planetary mission. Ranger VII achieved the first detailed observations of the lunar surface. Mariner IV has achieved the first successful mission to Mars.

It is a tribute to the 60,000 Americans working on the unmanned space systems that they have achieved a preponderance of significant firsts in space. This is particularly true since the Soviets have very active scientific satellite, lunar, and planetary programs. In fact, in some areas such as planetary flights, they have been more active than we, but with no successful missions.

Only in manned space flight have the Soviet space missions eclipsed our own. But here I also feel that we are in a position of great strength. I base this opinion on a knowledge of the truly tremendous preparations under way for Apollo. While much of the equipment is not yet flying, an outstanding team has been built and hardware progress has been excellent. It is hard for me to believe that we will not soon emerge the stronger in this area also.

All this breast-beating is not intended to belittle the Soviet effort. I have not mentioned their "firsts" in space, which are also impressive. They are capable, resourceful, determined, and in all respects worthy competitors in what I hope will remain a peaceful competition to expand man's knowledge about the universe he lives in.

CAN WE AFFORD A CIVILIAN SPACE PROGRAM?

This rhetorical question is usually asked to suggest that the dollars going into space exploration could be readily diverted to alleviate many of our earthly ills. Such is not the case, however.

Dollars invested in the space program actually represent an investment of people. About 380,000 people -- or one-half of a percent of our national work force -- participate directly in the civilian space program. Of these, there are 80,000 scientists and engineers who comprise about 6% of our national supply. This group is uniquely suited to pioneering in the physical sciences and in electromechanical technology, fields in which we lead the world.

Fortunately, history has shown that our greatest strides in improving human welfare come as adjuncts to generally unrelated progress in science and technology. Thus, my answer must also be rhetorical: can we afford not to explore space?

WHY HAVE BOTH A CIVILIAN AND MILITARY SPACE PROGRAM?

The answer here lies in the fundamental difference in the respective roles of NASA and the DOD. NASA's role is to explore and exploit space for peaceful pruposes. The DOD's role is to stay prepared to defend the United States and its allies, operating in any medium that furthers this end. The present space program with its great breadth would never have evolved under the DOD, which must necessarily devote its full attention to its awesome military responsibilities. Congress saw this clearly in drafting the Space Act of 1958.

Fortunately, the two space programs are mutually supporting and blend together quite well. They use common equipment in many instances, and draw on the same scientific and industrial base. In addition, numerous projects are of great mutual interest. Top management in both agencies devotes substantial effort to insure close cooperation and to minimize duplication. Public service must clearly override self service -- and does!

WILL THE UNMANNED PROGRAM BE SWALLOWED UP BY THE MANNED PROGRAM?

This will never happen because it would make no sense at all. It would be a step backwards, somewhat analogous to replacing guided missiles with manned aircraft.

Unmanned spacecraft have chalked up over 50 successful space missions including such pioneering flights as the Explorers, OSO, OGO, TIROS, Nimbus, Echo, Relay, SYNCOM, Pioneer, Mariner, and Ranger. There missions have clearly demonstrated the unique ability of the unmanned spacecraft to effectively extend man's sensors into space and to distant worlds, as well as to perform functions of great practical utility. In my opinion, we have barely scratched the surface.

At the same time, manned space flight is both inevitable and desirable. Man has unique capabilities as an explorer which are difficult if not impossible to duplicate with automated equipment. These include the ability to observe, reason, and modify plans in unpredictable situations. Once on the surface of the moon and planets, man will be the master of the machine. Furthermore, in earth-orbiting laboratories of the future, man will develop technologies seen only dimly now.

But for continuous monitoring of the space environment, for exploring the farthest reaches of space, for probing into unknown and hazardous regions, for providing year-in and year-out utility, for being the pathfinders for all space exploration -- unmanned spacecraft will remain without peer.

WHAT IS THE SINGLE MOST DIFFICULT PROBLEM IN CARRYING OUT THE SPACE PROGRAM?

Management: When I got into this business in 1958, I wouldn't have said this. My professional world consisted of a host of technical problems in need of solution. My prerequisites for a successful project were a sound initial concept, solid engineering, careful fabrication and craftmanship, exhaustive testing, and, above all, good technical men.

These are still prerequisities for a successful project. But the most critical element is missing -- management. Without first rate management, one or more of these prerequisities will fall short of the mark. Without first rate management, the system will fail to go together smoothly and function as a unit. Without first rate management, the project will simply bog down while time fleets and costs rise.

The culprits will be such things as cumbersome organizations, poor communications, improper delegations of authority, lack of supervision and attention to detail, low morale, carelessness, or any one of the host of possible management ills.

I could cite case after case in industry and in government where poor management was the root of what superficially appeared to be the normal technical problems of the research and development business. In an industry where perfection is required from the outset, the projects must be well managed down to the lowest tier subcontractors. If we have learned one big lesson since Explorer I, this is that lesson.

ARE INCENTIVE CONTRACTS JUST ANOTHER GIMMICK?

No. I believe they are here to stay -- and for good and sufficient reason. While the mission accomplishments of our space projects have been truly remarkable, our adherence to schedule and funding plans has left something to be desired. In addition, reliability and quality control has been a persistent problem. We felt we needed enlightened and aggressive company management to help solve these problems. We weren't getting it in many cases. A prime purpose of incentives is to insure this management attention. A secondary objective is to improve the performance of the government project teams by requiring better initial project definition and fewer changes during the course of the projects.

Incentive contracts have not been easy to work with in the R&D business. We have made mistakes in their application and will, no doubt, make more. But, despite these problems, we are learning and are beginning to see some positive results.

WHAT IS THE ROLE OF NUCLEAR ENERGY IN THE SPACE PROGRAM?

Radioisotope power supplies are useful today. They are particularly well suited to lunar and planetary landing systems where solar cell arrays can be a problem. Also, they can simplify and extend the life of some types of satellites.

Nuclear reactors in space are not required for any currently approved NASA missions. One day, however, they will supply the large amounts of electrical power required by direct braodcast communication satellites, unmanned planetary probes such as an advanced Voyager, unmanned probes far away from the sun, manned orbiting laboratories, advanced manned lunar expeditions, and manned planetary flights.

Electric propulsion will also require nuclear reactors but I do not expect a requirement for such systems until the 1980's. This may sound conservative, but I base my judgement on current planning, ability of existing systems to do the interim job, and the high cost and long development times of a fully integrated electric propulsion system.

The nuclear rocket has made great progress. It is fairly clear that it can be developed into an operational system. However, I believe the nuclear rocket will not be required for anything short of manned planetary exploration and will not get a full "go ahead" until that total mission is undertaken. Once developed, it can also serve a useful purpose in advanced lunar missions, such as a lunar base.

WHY CONFINE OURSELVES TO MARS AND VENUS WHEN THE WHOLE SOLAR SYSTEM IS ACCESSIBLE?

We are occasionally accused of being too conservative in this respect. We are, of course, fully aware of the accessibility of the solar system with relatively straightforward launch vehicle and spacecraft developments. Naturally, we would like to conduct missions to the Sun, Mercury, Jupiter, Saturn, and the outer planets, as well as to comets, asteroids, and out of the ecliptic. We have extensive studies of such missions underway.

Such an effort would be a large drain on our resources, however. Rather than spread our manpower and funds too thin, we have elected first to do a workmanlike job on our nearest neighbors - the Moon, Mars, and Venus. Project Voyager constitutes a major step toward the planets Mars and Venus. The others will have to wait their turn -- which will surely one day come, and the sooner the better.

WHEN WILL WE HAVE MANNED PLANETARY EXPLORATION?

Manned planetary exploration is inevitable, but it will be very, very difficult. In October 1958, I had the opportunity of making NASA's first public presentation of its plans at an IAS banquet in Washington. Under pressure, I speculated off-the-record that man might orbit in three to five years, fly to the moon in seven to ten years, and fly to Mars in fifteen to twenty years. The next day, I was embarrassed to see these estimates on the front pages. I really had very little confidence in them.

However, a lot of good men made an honest man of me on the first date, and may do the same on the second. Not on the third. I do not believe that manned landings will be made on Mars before 1985 or later. My reasons? The long trip time, the problems of duplicating the LEM feat through an atmosphere, and the low likelihood of funding such a costly mission until after successful completion of Apollo.

HAVE WE COMPROMISED OUR SCIENTIFIC INTEGRITY IN "SELLING" OUR PROGRAMS TO CONGRESS?

This recent question went over like the proverbial lead balloon in my office. To give the devil his due, however, I think it is a fair question to ask. There are certainly many pressures which could lead to such compromises.

The answer, however, is a loud "No!" Integrity is insured by placing leadership in the hands of trustworthy men. It is important to recognize, however, that the leaders of our space program are expected to make decisions. They go to the Congress with the best programs they can devise after hearing from all sides and weighing all the arguments. They do not present a shopping list with all the pros and cons and ask Congress to make the decisions for them. Where their judgement is subject to doubt, I have never found Congress reluctant to ask for more information.

DO MANY SCIENTISTS REALLY OPPOSE THE SPACE PROGRAM?

No, most of them support it. The hundreds of scientists actually conducting flight experiments in the space sciences program are, of course, enthusiastic supporters. Thousands who are not directly involved are also strongly behind the program. I believe they recognize that the space program is a terrific shot-in-the-arm for science, engineering, and education in this country. Anyone who doesn't believe this need only talk to a typical youth of today.

Some opposition, however, comes from sincere men on intellectual grounds. They may object to the way some aspect of space science is being handled, or they may object to the emphasis on space science vis-a-vis other branches of science.

The only opposition that is really annoying is that based on self service or on misinformation. In one recent critique which made the front pages across the country, the scientist certainly failed to apply the scientific method. He criticized a major new project for confining itself to a particular objective and neglecting other areas which, in his judgement, held a higher probability of scientific return. All of our public announcements, however, had clearly stated that his recommended objectives were an integral part of the program. When asked, he was unaware of this fact.

Fortunately, these problems are relatively rare. Constructive criticism is healthy and we take it very seriously. We are particularly attentive to criticism from the scientific community which renders such invaluable service to the program, and which, in turn, is served.

N66 31504

MANNED PLANETARY RECONNAISSANCE

MISSION STUDY: VENUS/MARS FLY BY

By

H. O. Ruppe

NASA - George C. Marshall Space

Flight Center

INTRODUCTION

It appears logical to assume that manned interplanetary missions will be performed no earlier than well after the successful completion of the Apollo project. Under this assumption, this study investigated the feasibility of performing manned Mars and Venus flyby missions using available hardware; modified, where required. The following areas were emphasized:

(1) Conceptual design of a spacecraft with a high inherent reliability.
(2) Development of a reasonable mission profile.
(3) Identification of developments beyond Apollo/Saturn V technology which are required.
(4) Development of a reasonable time and cost schedule.

Interplanetary stop-over missions have been discussed in the literature quite long ago. Ziolkowsky gave thought to them. Hohmann put flight mechanics and astrodynamics on a firm basis (Hohmann: Die Erreichbarkeit der Himmelskoerper, 1925); Oberth improved on his work (Oberth: Wege zur Raumschiffahrt, 1929), and von Pirquet indicated a line of thought which led to the fast missions (von Pirquet, in "Die Rakete," June 1928). Pre-Sputnik, the system analysis by W. von Braun (von Braun, Das Marsproject, 1952) should be mentioned. Breakwell, Dixon, Dugan, Ehricke, Gillespie, Himmel, Magness, Lawden, Luidens, Ragsac, and Ross are some names that come to mind as authors of more recent works. Electric propulsion was discussed earlier by Oberth; Stuhlinger, Irving, Michielson, Moeckel, Pinkel, Edelbaum, Zola, Sauer, and Melbourne are more current investigators of that subject.

Because of the difficulties of these manned planetary missions, much thought has been given to ways of alleviating the problems; unfortunately, this usually introduces new ones. This brings to mind the investigations into mixed high/low acceleration systems (Edelbaum, Leovy, Widmer); the usage of high speed aerodynamic braking maneuvers both at Mars and Earth (MSFC, MSC, STL, Lockheed); the development of new mission profiles (Kaempen, Fonseca); or the development of more powerful means of propulsion (Orion, Fusion). Knowledge of this information is helpful to put the flyby mission into proper perspective, and to really get a feel for its relative simplicity.

The existence of interplanetary flyby trajectories (which originate and terminate at Earth, using essentially no propulsion enroute) was discovered quite recently (1956) by Crocco. Crocco's presentation to the Congress of the IAF in Rome, Italy initiated thinking in that area, and H. O. Ruppe developed an essentially complete survey of the various types of interplanetary round-trip trajectories, published in the Handbook of Astronautical Engineering. Many of the studies of this field are trajectory oriented (Gedeon, Ross), but system studies have been performed for both unmanned (Battin) and manned (EMPIRE) vehicles. Some special cases have been examined; noteworthy are the multiplanet flyby trips or Grand Tours of Space (Crocco, Dixon, Ross, Minovich); the combination of landing/flyby missions (Fonseca, Titus, Faget); the out-of-the-ecliptic flight (Breakwell); and a potentially promising type of unmanned solar probe (Ross).

One of several impulsive propulsion maneuvers could be applied enroute during the flyby trajectory, leading to a so-called "propulsive flyby trajectory." Two basically different cases have to be considered:

(1) The propulsion is used only to slightly modify a non-propulsive flyby trip, e.g., to make up for a launch delay.

(2) The propulsion permits the flying of a new type trajectory.

For case 2, only a few solutions are known: The Hohmann round trip of one and one-half years duration, published in 1925, and some generalizations thereof, and a modified Crocco round trip, which was recently discovered by H. O. Ruppe (to be published in Astronautics, Academic Press). Mr. W. A. McRae of North American Aviation, under contract NAS9-3499, found an interesting example using perihelion propulsion. For case 1, a report by Gobetz (A110058-5, Optimum Transfers between Hyperbolic Asymptotes; UA Research Laboratory, November 1962) covers very well the problems of optimization of the propulsive encounter. This work has been extended and applied to the flyby case in NASA contract NAS8-2469, phase III, Lockheed/Interplanetary Transportation Systems.

A milestone was the symposium on "Manned Interplanetary Mission Studies," MSFC, January 28-30, 1964, summing up the knowledge in this area at this time (edited by J. N. Smith, and published in NASA TM X-53049, June 12, 1964).

This report summarizes the study efforts performed at MSFC during the period August 1963 through November 1964.

MANNED FLYBY MISSION JUSTIFICATION

After Project Apollo has been completed in the latter part of this decade and man has returned safely to Earth after landing on the moon, the preparation for the next step in the exploration of our solar system must be well underway, to preserve continuity and momentum.

What the next step will be is not known at this time. The concentration of effort will probably be on a permanent space station, or on a lunar base, or possibly on the most ambitious mission within the current state of the art; namely, manned planetary flights, to Mars or Venus. Most desirable is a proper mixture of these activities, as demonstrated by the MSFC inhouse "Evaluation Procedure for Alternate Program Plans," as developed and utilized under the direction of the Future Projects Office.

The Apollo Project, as well as a lunar base or a space station, are not the end product, but are intermediate steps in the logical and systematic overall exploration of space. These three ventures are still within the Earth's "sphere of activity," and could be considered as training and development for the more ambitious missions of manned planetary landing, besides the inherent value of their own.

It is inconceivable that, with the tremendous manpower, facilities, and technologies being developed, manned planetary missions will not occur. Let us assume that a manned landing on Mars is to be performed. Then one or more of the following have to be developed:

(1) Large Post-Saturn launch vehicle
(2) Solid-core nuclear propulsion
(3) Hyperbolic aerodynamic braking capability, Earth
(4) Aerodynamic braking capability, Mars
(5) Earth orbital operations
(6) Refueling at Mars or the Moon
(7) Electric propulsion
(8) Nuclear pulse propulsion
(9) Gas Core Nuclear Propulsion
(10) Fusion rocket engine

A manned Mars landing and return mission, using all chemical propulsion, requires 3 to 10 million pounds in Earth orbit. When you consider that the pauload in Earth orbit for a Saturn V is about 250,000 pounds, such a mission seems highly impractical. The availability of a Post-Saturn launch vehicle with a payload capability of about 1,000,000 pounds would alleviate the orbital operations burden, but the mission would still be difficult.

A significant reduction of initial mass in Earth orbit is possible if we can use aerodynamic braking at Mars or refueling there, but these methods assume a knowledge about either the composition, temperature, pressure and density profile of the Martian atmosphere, or about Mars surface resources which just is not available and, in light of the anticipated flight schedules, will not likely be forthcoming before the mid-1970's from unmanned probe activities. In addition, testing for this aerodynamic braking will be a necessity, and at the target planet, at least for demonstration pruposes.

The most reasonable way to create a planetary exploration capability seems to be the development of propulsion more exotic than chemical. The leading contenders are solid-core nuclear, electric, and nuclear pulse, with the solid-core device best in hand and probably available at the earliest date.

Assuming, then, that the pacing item for the manned Mars landing mission is the availability of solid-core nuclear propulsion, which -- with adequate thrust -- might not be man-rated and operational until around 1980, we have ruled out such missions during the 1970's.

To design and develop, during the next decade, systems for a manned Mars landing mission in the early or middle 1980's, with so many unknowns, appears extremely difficult. As an example of this difficulty, consider: How do you design a Mars landing module to cope with the range of surface winds that are postulated? The unknown surface features? The variation in predicted surface pressure? Atmospheric composition and temperature?

The probable method to overcome these unknowns would be to design for "worst conditions" and hope these are not exceeded. But this could become very costly, not only in funds but also in total weight. Indeed, it is conceivable that by this procedure, systems would be developed and built which are not needed at all.

The first venture, still assuming that we are not very knowledgeable about Mars' properties, would probably transport 2 or 3 men to the surface of Mars, for a few days. This leads to a cost on the order of a billion dollars per man-day on Mars.

If the physical properties of Mars were well known, we could think in terms of the first landing as a long-duration base, reducing cost to less than 10 million dollars per man-day on Mars. The benefits of aerodynamic braking, as another example, cannot be realized unless the Mars atmospheric properties are known.

The question is: How can this information be obtained? The most apparent answer is unmanned probes. But the scheduled Mars probe program is meager. However, unmanned probes have very useful features, e.g., they are lightweight and do not need return capability.

Let us consider combining these advantages with the best features of manned missions, i.e., checkout, maintenance, and repair capability. This is done most easily with the minimum manned planetary mission, namely, reconaissance flyby. In this mission profile the spacecraft performs a close approach hyperbolic encounter with the target planet, then returns to a rendezvous with Earth. The only major propulsive maneuver is at Earth escape.

Just before and during passage of the target planet, many probes, which have been carefully maintained during the trip, are released toward the planet. Some would land, some orbit, some could perform aerodynamic braking tests, and some float within the atmosphere. The spacecraft would monitor these probes for many days and relay the information back to Earth. The short transmission distance between probe and spacecraft is advantageous. At a later time, when that distance gets too large, the probes might transmit to Earth, at a lower bandwidth.

The following landing missions would profit also from the crew training as well as from the experience with many operational areas, such as launch window utilization, Earth recovery, midcourse guidance procedures, zero-g procedures, etc. In addition, complete subsystems might be transferred to such later missions, as e.g., the life support system, astrionics and communications subsystem, etc.

Although many gains can be realized from the flyby missions, the major advantage is that these missions can be performed using hardware presently under development: The Saturn V for Earth surface launch; modified S-II or S-IVB stage as the orbital launch vehicle; and the Apollo commond/service module for major elements of the spacecraft. No new engines would be needed. The only new development would be the living module, and that might be adopted from the orbital program.

There is too much talk about the Apollo project being a "dead end." With proper planning, this is not so -- orbital and lunar missions will utilize this hardware and experience. With an additional customer for this hardware, such as a manned planetary mission, the "cost effectiveness" would be further improved. Thus the flyby missions will be an excellent opportunity to "cash in" on the Apollo investment, and open the door for a follow-on program.

Another point that should be mentioned is "prestige and pride." From the lunar landing in this decade to a possible planetary landing in the early or middle 1980's is 10 to 15 years. Without a major new undertaking, public

support will decline. But by planning a manned planetary mission in this period, using developed hardware, the United States will stay in the game, and at the same time perform a logical step in the systematic exploration of space, keeping an option for the "Man-on-Mars" project.

GENERAL MISSION DESCRIPTION

It is felt that a general description of the missions will improve the clarity of the subsequent discussions.

Mars Mission

The first flight from Earth surface puts the unmanned interplanetary spacecraft, an S-II stage, and the adapter between the spacecraft and the S-II stage into a 185 km orbit. The S-II stage separates and is jettisoned. The remaining total mass is between 183,000 pounds (Venus, 1978) and 274,000 pounds (Mars, 1973). The adapter contains the Saturn V instrument unit and a propulsion system (two RL-10 engines with spherical tanks, some low thrust storable propellant engines for vernier maneuvers and attitude engines) which is used to rendezvous the spacecraft with the S-IIB stage. Actual docking might employ the attitude control engines. The spacecraft mass is about 200,000 pounds; its size is comparable to the S-IVB stage. Four LOX tankers are required to transport the required LOX (up to 772,000 pounds) to the S-IIB, and are put into the 185 km orbit.

Lastly (in order to minimize the staytime of liquid hydrogen in space), the orbital launch vehicle (modified S-II stage -- referred to as S-IIB) is put into a circular 485-km altitude orbit. The vehicle contains up to 155,000 pounds of liquid H_2. (The total Saturn V payload capability to a 485-km orbit is assumed to be 257,200 pounds; the nominal standard Saturn V capability to that orbit is about 220,000 pounds.) The crew (three men) board the spacecraft in the 185 km orbit, and both spacecraft and tankers rendezvous with the S-IIB. Finally, launch from the 485 km orbit occurs, using the S-IIB stage.

The S-IIB stage is separated and jettisoned. The RL-10 engines in the adapter might be used at this time for injection vernier; then the adapter is jettisoned. If advantageous, however, the adapter could be left attached to the spacecraft, e.g., for meteoroid protection of the spacecraft aft end.

The spacecraft proceeds along its trajectory with continuous attitude control, but with no simulated gravity. A simulated gravity version is more complicated and heavier. Midcourse propulsion is provided by a Lunar Excursion Module descent engine (thrust: 10,000 pounds).

During the Mars encounter, 10,000 pounds of scientific probes are put on and around Mars. The probes are then monitored by the spacecraft and data are relayed back to Earth. Later, the probes might communicate with Earth at a reduced bandwidth.

Some days prior to Earth encounter, the crew will transfer into the command module to perform checkout adjustment and repair operations. At this time, the three-man crew switches to the Apollo life support system. Well before Earth return, both command and service module are separated from the hangar, and positioned for Earth retro-braking. Prior to atmospheric entry, the service module is utilized to brake the speed of the command module from its arrival condition to Apollo design conditions (parabolic speed). The service module is jettisoned thereafter, and the command module performs the normal Apollo entry and landing maneuvers.

Figure 1 summarizes the interplanetary mission profile.

Venus Mission

The Venus mission is essentially identical to the Mars mission. The same orbital launch vehicle (S-IIB) is used except for being off-loaded and having possibly a correspondingly reduced length of the cylindrical section of the H_2 tankage. The spacecraft is also the same except for off-loading due to less rocket retro-braking, reduced attitude control, and reduced life support system requirements because of the shorter mission duration.

A summary of the speed requirements follow; to these numbers corrections for gravity losses, Vernier corrections, and performance reserves have to be

added (a total of, typically, three percent).

	Launch Speed (ideal) from 485-km Earth orbit, km/sec	Earth Arrival Speed above parabolic, km/sec
Venus	3.8 ± 0.2	2.6 ± 0.3
Mars	5 ± 0.2	4.1 ± 0.7

INTERPLANETARY SPACECRAFT

Figure 2 shows the layout of the interplanetary flyby spacecraft configuration. This configuration is designed for zero gravity. Equipment for physical exercise and a centrifuge are provided within the crew living area. The spacecraft is adaptable to either the Mars or Venus mission with the only changes required being the adjustment of the amount of midcourse and braking propellants and life support mass. For descriptive purposes the spacecraft is divided into four major segments:

(1) Aft Skirt Assembly

The aft skirt assembly consists of a conical adapter section which houses the Earth orbital transfer propulsion system and its propellants, docking structure for Earth orbit rendezvous, and docking and attitude control propulsion systems. It is assumed that the 3,500-pound instrument unit, located in the aft skirt, is the "brain" for attitude control and rendezvous maneuvers. The propellant tanks and pressurization system which supply the docking and attitude control system are centrally located between the crew living quarters and the hanger. They feed the engines in the aft skirt through disconnectable lines. The aft skirt assembly is jettisoned with the spent injection stage after the spacecraft is boosted into a transplanetary trajectory.

(2) Laboratory/Crew Living Area

The laboratory/crew living area was designed to accommodate three astronauts and their life support system for the time required to complete a Venus or Mars flyby mission. Environmental conditions for the crew living area are discussed later. The crew area is a spherical shell with an outside wall diameter of 240 inches. The exposed outside hemisphere consists of an aluminum inner wall 0.130-inch thick, two meteoroid bumpers of 0.025-inch thick aluminum (per bumper), and foam insulation separating the aluminum walls. Since the inner hemisphere is within the cylindrical section of the hangar, it is constructed with one inner wall of 0.130-inch thick aluminum, a single meteoroid bumper of 0.025-inch thick aluminum, and foam insulation separating the two walls. A cylindrical solar flare shelter is located inside the living area and is protected by the various life support subsystems, which are arranged around the shelter. The shelter contains sleeping quarters and emergency controls. The walls of the shelter are water-jacketed and contain the spacecraft water supply system.

(3) Apollo Hangar

The Apollo command (for reentry) and service (for rocket braking) modules are stowed in a hangar section at the front (before injection) end of the spacecraft. The construction of the hangar is similar to the laboratory. The two Apollo modules are supported inside the hangar by conical supports which connect to the bulkhead sections of the braking module. Emergency provisions, various life support systems, and the probes to be dropped at the target, are located in the hangar. The equipment, in both the laboratory and hangar sections, is mounted on panels away from the shell to allow rapid leak detection and repair. An airlock tunnel connects the hangar and the living area. The radioisotope power supply systems and midcourse correction engine are also located on the front (before injection) section of the spacecraft, protected during ascent from Earth by a nose fairing. The power supply systems will be extended by the astronaut in orbit, long after the fairing has been jettisoned during boost from Earth.

(4) Midcourse Propellant Bay

It is assumed that continuous pressurization of the cylindrical section between the laboratory and hangar is not required. Therefore, the outer cylinder is not designed for meteoroid protection, but on a structural basis only. This results in a minimum skin thickness of 0.091 inch. For this

reason, all storable propellant tanks (midcourse, docking, attitude control), pressure vessels, and related equipment located in the cylindrical section have been provided with supplementary meteoroid bumpers. The storable propellant tanks located in the cylindrical section are shown in Figure 2. In the event of an emergency requiring an astronaut to enter this area for a short time, the pressurant required to pressurize this area would probably be less than the structural weight required for meteoroid protection. If the astronaut can wear a spacesuit, such pressurization would not be required.

Design Ground Rules

The basic ground rules which governed this design are:

(1) The spacecraft is to make use of existing Apollo hardware and equipment to the greatest possible extent.

(2) The Apollo command module and service module and the probes are to be enclosed in an environmentally controlled hangar, to protect them against micrometeoroids, outgassing, and other detrimental effects.

(3) A three-man crew is assumed using the Apollo reentry vehicle.

(4) The laboratory/living area is to be separate from the Apollo hangar area.

(5) All personnel are to be housed in the laboratory/living area.

(6) Solar flare protection is to be provided by a storm shelter throughout the mission.

(7) The laboratory/living area is to be environmentally controlled to a shirt sleeve environment using a mixed atmosphere of nitrogen and oxygen.

(8) An air lock is to be placed between the hangar and laboratory areas.

(9) Both the laboratory area and the hangar area are to be equipped with life support equipment if either becomes temporarily uninhabitable.

(10) The Earth reentry velocity is to be 11,000 m/sec at an altitude of 120 km.

(11) The spacecraft is to be completely sealed after entry of the astronauts to minimize leakage. Therefore, there should be a low probability for the astronauts to go outside the spacecraft to perform maintenance. If required egress and entry occurs through an airlock from the hangar.

(12) Midcourse and braking propellants are to be Earth storable hypergolics.

(13) Approximately 10,000 pounds of mass is allotted to the unmanned scientific probes.

(14) Command module reentry weight is assumed to be 9,790 pounds.

(15) Injection into the planetary transfer trajectory is from a 485-km Earth parking orbit.

(16) Life support atmosphere storage is sized for 12 repressurizations (to flush trace contamination to space) for the Mars mission and 8 repressurizations for the Venus mission.

(17) The present service module, having a propellant capacity of 45,000 pounds, is used for Earth return braking from arrival speed to parabolic speed, adjusting its propellant mass accordingly.

(18) The following atmospheric conditions are to be used in determining the life support requirements:

Pressure - 10 psia

Temperature - 70°F

Relative Humidity - 50%

Composition - 50% O_2 and 50% N_2

Concentration of CO_2 - 0.5% maximum

Leak Rate - 5.0 lb/day (total spacecraft).

(19) Continuous attitude control is maintained during the mission.

Internal Power Supply

For reliability reasons, thermoelectric/isotope power systems are used. There are 20 units, having an output of 0.5 km (electric) each. The total weight of this system is assumed to be 10,000 pounds. This weight may be high, however, for two reasons; first, the total power of 10 kw(e) may not be required; second, RCA is at present studying a generator for lunar surface use

of 50 watts output with a weight of less than 30 pounds, which would lead to a total weight of only about 5,000 pounds for a 10-kw system.

Radiation Shelter

For solar flare protection, a radiation shelter is provided. The study by the Research Projects Laboratory of MSFC shows that for these missions in the latter half of the 1970 decade no shelter is required, thus saving approximately 1,000 pounds of spacecraft weight. Since this represents only about 0.5 percent of the overall spacecraft mass, the shelter was retained as an additional safety feature, rendering the spacecraft useful during times of larger solar activity.

Astrionics Systems

For Earth orbital operations and injection, the instrumentation unit is utilized. Thereafter, we have the astrionics system on board the Apollo command module. In addition, there is a total weight of 4,500 pounds allowed for displays, communications, instrumentation, guidance, and navigation, plus other astrionics equipment. This weight is on top of the internal power supply, but does include systems such as special environmental control, cables, batteries, computers, power conditioning, spare parts, etc.

(1) Ascent into Earth Orbit

The launch from Earth into an orbit will use the ST-124 stabilized platform along with the other Saturn V astrionics. If the stay-time in the 185-km orbit exceeds one revolution, ground-based tracking may be used to update position and velocity knowledge in the vehicles and essentially eliminate the errors from the ascent portion of the mission. If uncorrected, the drift of the gyros will cause the platform alignment error to increase as the vehicle stays in the 185-km orbit for phasing. The requirements for updating of the alignment of the platform should be investigated for the boost phase from the 185-km orbit to 485 km.

(2) Rendezvous

The guidance equipment required for the rendezvous will be an inertial reference augmented by radar and possibly optical equipment. The degree and implementation of manned participation in the docking phase needs investigation.

(3) Manned Interplanetary Flight

The subsystems that are required in these phases of the mission should be a basic set, augmented by phase peculiar items. For instance, an inertial reference will be required for powered phases, as will an on-board computer. The use of on-board radar (or laser) may be desirable at close approach to Mars and for the return and braking at Earth.

The need for attitude stabilization of the vehicle will exist periodically for communications with Earth and for on-board measurements to be used in navigation. Otherwise, the vehicle could be allowed to tumble to conserve attitude control propellants. Since the penalty is small, in this study continuous attitude stabilization -- controlled by star trackers -- was assumed.

The midcourse correction requirements will be a function of the injection uncertainties, lack of knowledge of astronomical constants, and unknown or unpredictable forces acting on the spacecraft during the free fall phase. Presently there is an allocation of 500 meters per second for midcourse correction. This magnitude may change as definition of the forces acting upon the vehicle is made. The programmed venting and the leakage of air acting over the period of one to two years may give sizeable velocities that either could be utilized to advantage or that must be removed by midcourse corrections. The solar wind, meteorite impacts, and radiation pressure may contribute to the required magnitude of midcourse velocity. A study will be required to determine a reasonable velocity reserve for midcourse corrections.

(4) Midcourse Navigation

The midcourse navigation may be done by ground based computers and sensors (DSIF) or by on-board measurements and computation. The use of ground-based tracking and navigation has been previously studied, is described in several reports, and has seen application with Mariner II and IV.

The problem of on-board navigation of a vehicle in space may be divided into three major areas. First, one must measure those observables which are related to the state of the vehicle. Second, these measurements must be used

in performing calculations which will give the state of the vehicle. Third, the present state must be compared to the anticipated or desired state in order to determine whether or not one is on course. If not, corrections to the course must be made. These three major areas will be discussed more fully in the following paragraphs.

In measuring the observables, the problem of instrumentation arises. It is found that the basic observables do not depend on the mission, nor does the general type of instrumentation available to take data on these depend on the mission. The mission dependence comes in the way the measurements on the observables are used in the second phase of the navigation problem. Of course, some observables may be easier to use than others for a particular mission, but the field of available observables is constant for all missions.

The instrumentation that may be used for measurements on the observables can be implied by examining the observables to be used. For operations near a celestial body, star sightings, landmarks on the body, stadiometric sights, occultations, and direct relative velocity and range measurements are probable. These imply theodolites of various types including sextants, telescopes, radar or similar devices, a timer or clock, and accelerometers and/or platforms for dead reckoning.

In deep space we must eliminate, from the above, direct velocity and range (stadiometric) measurements, as the celestial bodies (except the sun) will be, for all practical purposes, point sources of light. Thus, we are reduced to using only explicit and implicit triangulation techniques plus dead reckoning. The instrumentation will, therefore, be only theodolite-type devices* and a clock, with possibly a platform and accelerometers.

In determination of the state of the vehicle we may follow two courses. First, we may use explicit navigation or second, we may use implicit navigation. Explicit navigation is performed by taking the measurements on the observables and reducing them directly to position and velocity information. The advantage of this technique is that it provides the navigator with direct data on vehicle position and course. The disadvantage of this technique lies in the need for an elaborate computation capability, especially in deep space missions.

Implicit navigation does not result in a determination of the present position. Instead, a set of conditions are examined to determine whether they are met or not, and if they are not met then the amount of deviation from the desired set of conditions is a measure of deviation from the desired course. The best way to visualize this type of navigation is to think of a vehicle following a well defined reference trajectory with only slight deviations from this trajectory occurring. The validity of implicit navigation techniques depends upon the constraint of small deviations from the reference.

The choise of explicit or implicit navigation must be made after a study of the mission, a choice of emphasis on either on-board or ground-based capability, and an examination of the instrumentation required to implement each of the techniques.

Generally speaking, a long duration, manned deep space flight should have as a prime navigation mode an implicit approach. Explicit techniques should be available for near-body operations, and possibly as a backup to the implicit technique in the event of unplanned excursions from the standard trajectory. Use of the instrumentation and computational capability currently considered for manned missions to implement the navigation scheme as described above would result in a less accurate deep space capability in the explicit mode, but near-body operations would be of the same accuracy of either mode.

In comparing the state as determined implicitly or explicitly with the desired state, and in deciding on whether or not a course correction must be made, it is necessary that the navigator be able to determine the furute state from a knowledge of his present state. Again, we must examine this problem in two ways, implicitly and explicitly. In the explicit mode the present state is used as a set of initial conditions to the equations of motion, and then these equations are a set of initial conditions to the equations of motion, and then

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*A *telescope coupled to a camera to photograph planets, etc., against the star background, plus man and a measuring microscope to evaluate the photograph, promises to be a lightweight, simple, reliable, and accurate system. It will be slow, which may be of no particular disadvantage here.*

these equations are integrated to some later time in order to find the future state. This obviously would require an elaborate on-board computation capability. A simplified explicit technique takes advantage of a pieced conic approximation to the trajectory. In this approximate method, conics are chosen that fit a section of the trajectory fairly closely, and the vehicle is then flown along this path until such time as a new conic is required in order that the vehicle does not deviate too much from the true trajectory. Although this approach requires the same sort of preknowledge of the mission as the implicit technique for implementation, it is simpler than the explicit technique in terms of computational capability required, and it is simpler than the implicit technique in requirements on the memory portion of the computer.

In summary, it must be pointed out that the major factors influencing the choice of a navigation scheme are: (1) Examination of the mission to determine the requirements to be imposed on the on-board capability. That is, is the system to be operable in both manned and unmanned modes? Will the dependence on ground-based capability be high or low? Do other requirements on the mission imply extremely accurate navigation during certain phases or not? These and many more questions must be answered before a choice of navigation scheme can be made. (2) Once the mission factors are settled, those observables best suited for use during various phases of the mission must be chosen, and then the instrumentation for obtaining data on these selected. (3) Through considering the mission requirements and the instrumentation which can be allocated to navigation, a choice of explicit versus implicit navigation may be made.

(5) Communications

The extremely long range (Mars: up to 3.2 A.U.), and weight, power and volume limitations present a challenging problem. Extreme care in design to provide inherent reliability and adequate repair capability must be planned well in advance of the mission.

In most studies of space communications, the first step is to select the optimum transmitter frequency for the spacecraft to Earth link. This is usually based on the free space attenuation, antenna gains, and galactic or background noise. The lunar links and narrow bandwidths of space probes usually optimize around 2, 300 mc. Using these criteria for the ranges to Mars, the frequency goes up to 8,000 or 9,000 mc because of the enormous space loss.

One of the ground rules for this study was that existing equipment would be utilized wherever possible. Therefore, frequencies compatible with the projected DSIF and Apollo networks were assumed, i.e., 2,295 mc. The characteristics of the DSIF for the 1975 time period were used since data on the manned space flight network were still tentative.

The weight of the ancillary equipment in pounds is tabulated below:

Communication Receiver and Decoder	70 lb
TLM and Signal Conditioning	80 lb
Timer and Clock	20 lb
Data Storage	180 lb
Television and Optics	200 lb
TOTAL	550 lb

Assuming 600 watts total power consumption for the communication system, of which 100 watts are vehicle antenna output, we obtain for bandwidth: 20,000 cps during Mars encounter; 100 cps during "worst condition" (Vehicle - Earth at conjunction, with the Sun nearly between them; with the Sun between vehicle and Earth there will be no communication for several days). What can be done with such performance? The higher value gives five voice channels, or one very low quality TV channel in nearly real time, or five high quality color pictures in three hours. The lower value permits one teletype channel, or one synthetic language speech channel.

(6) Earth Return/Landing

The Earth return and landing proceeds in essentially five phases:

(a) Place the spacecraft on the proper return corridor. This is not too critical since powered phase (c) can make up for small residual errors.

(b) Crew checks out and enters command/service module, and abandons spacecraft by separating command/service module.
(c) Ignite service module for retro propulsion from arrival condition to Apollo entry condition. (The service module is modified over the lunar version in that there is a different propellant loading and there is only one burn, which permits burst valves for the long duration storage.)
(d) Separate service and command modules.
(e) Command module performs Apollo-type entry and landing.

The braking engine delivers a thrust of 21,900 pounds, with a specific impulse of 313 seconds; therefore, the four above weights correspond to the following burn durations:

Venus

346 sec (1978); 481 sec (1972, 1980)

Mars

639 sec (1980); 1,166 sec (1973)

The present design specification for this engine calls for a maximum burn-time of only 630 seconds. To overcome this problem we must -- at least for the Mars missions -- either modify the engine to permit longer burn duration, or increase the thrust to reduce the burn duration, or we must install two engines in the service module.

(7) Reliability

A matter of great importance is the fact that there is not an astrionics system currently available which is suitable for a manned interplanetary vehicle. The functions and lifetime requirements are vastly different from those on any present, or soon to be available, system. These requirements manifest themselves in such items as high accuracy, low power consumption, simple operations, light weight, small size, dependable operations during a two year period, repairability, and ability to withstand the launch and free flight environment. The design of an astrionics system which will meet these new requirements will pose some formidable problems.

When the mission requirements are firm, a simple system should be postulated which will be functionally adequate, and an attempt made to predict the probability of successful operation of this system. When this is done it is expected that attention will be focused on many aspects of the design which will need additional work. It is foreseen that work will be needed in at least the following areas: Components, such as optics, inertial sensors, and transducers, if not available, will have to be developed. Use of redundant components and subsystems will have to be considered. Current techniques for establishing electrical inter-connections may have to be improved. A system that is repairable by using replacements will need to be considered. Systematic testing and evaluations of hardware items -- possible using a space station -- will be needed in order to obtain data suitable for a relistic prediction of the probability of successful operations of the system.

One conclusion which may be drawn from the foregoing is that it may be necessary to start work now on the astrionics requirements in order to insure an adequate system being available in 1975.

(8) Astrionics Weights

The present allocation of the astrionics equipment weight for the spacecraft is 4,500 pounds, not including power source or operating expendables.

Several factors can exert major influences on this weight estimate. Perhaps the most important is the basic reliability of the equipment and the consequent decision as to whether redundancy or in-flight repair philosophy is used to assure adequate system operation. It is apparent that even with major decreases in present day equipment failure rates at least one of these two methods will be required to assure high probability of successful system operation.

The environment control and duty cycle requirements have a direct effect on system weight and reliability also. There are many individual components that will have longer life if left operating or controlled to operating temperature, even though the system with which they are associated is required

only intermittently. This mode of operation must be traded off against the weight consumed in power generation, and against, where applicable, the reliability gain resulting from "shutdown periods."

Life Support System

Both an open loop life support system and a semi-closed system have been investigated. At the mission time frame being considered, the semi-closed loop should be available. Therefore, only the semi-closed system is discussed in this brief summary.

With the living module and hanger pressurized to 10 psi (50% O_2/50%N_2), and the partially open life support system, the spacecraft has the following weight losses:

Leakage (estimated)	5 lb/day
Repressurization	1,885 lb/event
Food, atmosphere, water and charcoal consumables	5 lb/man-day

Further life support systems weights are:

	Mars (lb)	Venus (lb)
Environmental Control System	1,000	1,000
Water Reclamation System (2):	500	500
Molecular Sieve:	300	300
Crew and Crew Support Equipment:	3,000	2,000
Hangar; Support & Maintenance:	1,000	1,000
Storm Shelter Support Equipment	500	500
Life Support Emergency Provisions	1,350	1,150
Life Support, etc., within Apollo Module for Final Phase of Earth Return	2,000	2,000
TOTAL	9,650	8,450

Structures

The weights (in pounds) for the spacecraft structure, aft skirt assembly and propellants follow:

Spacecraft:

Hanger	6,950
Cylindrical Intersection (Midcourse Propellant Bay)	3,100
Living Area	4,150
Shelter	1,250
TOTAL	15,450

Aft Skirt Assembly:

Docking Structure/ Orbital Transfer Propulsion System	14,876
Instrument Unit	3,500
Attitude Control System	500
Subtotal	18,876 lb

S-II Stage Addition, for Spacecraft Docking, usually counted in with the air shirt assembly	5,700
TOTAL	24,576 lb

Low Orbit to High Orbit Transfer Propellants:

O_2/H_2 - Stored in Aft Skirt Assembly	10,000
Storables - Stored in midcourse propellant bay, fed to aft skirt engines through quick disconnect lines	12,000
TOTAL	22,000 lb

Spacecraft Mass Table

Following is a weight breakdown of the spacecraft for a typical Mars and Venus mission:

	Mars (1978) 691-Day Mission	Venus (1974) 394-Day Mission
	(lb)	(lb)
Environmental/Crew Systems	9,650	8,450
Leakage	3,455	1,970
Repressurization	22,650 (12)	15,050 (8)
Food, etc. (Consumables)	10,365	5,910
Command Module	9,790	9,790
Crew Access/Docking Provisions for Crew Boarding (in hangar)	430	430
Service Module, Dry, and Support	10,894	10,833
Propellant, Service Module	47,790	29,567
Midcourse Propulsion, Dry & Pressurization, etc.	3,513	3,370
Midcourse Propulsion, Propellant (storable)	26,272	20,583
Attitude Control, Dry	500	500
Attitude Control, Propellant (2.5 lb/day)	1,728	986
Scientific Equipment on Board	1,000	1,000
Probes to be Dropped at Target Planet	10,000	10,000
Astrionics (additional to Apollo System in Command Module)	4,500	4,500
Internal Power (10 kw (e))	10,000	10,000
Structures	15,450	15,450
Contingency	1,942	1,942
TOTAL	189,929	150,331

For the Open Life Support System, Add:	21,000	12,000
For the Simulated Gravity Version Add:	69,000	69,000

Thus the following summary table results:

	MARS		VENUS	
	Closed	Open	Closed	Open
Zero Gravity	190,000	211,000	150,000	162,000
Simulated Gravity	259,000	280.000	219.000	231,000

Interstage to S-IIB, dry: 24,576 lb

The following table gives the extreme ranges of spacecraft mass encountered during the metonic cycle, for the zero gravity closed ecologic cycle version:

	MARS		VENUS	
	1980	1973	1978	1972/80
Spacecraft	186,046	230,092	143,707	154,782
Interstage	24,576	24,576	24,576	24,576
Orbital Transfer Propellant	22,000	25,000	20,000	22,000
SUM	232,622	279,668	188,283	201,358
Minus 5,700 lb, Carried with S-IIB	5,700	5,700	5,700	5,700
TOTAL to 185-km orbit (lb)	226,922	273,968	182,583	195,658

Since the injection speed requirement is not exactly constant from opportunity to opportunity, the heaviest spacecraft does not necessarily lead to the heaviest launch mass in orbit. Indeed, for these extremes:

	MARS		VENUS	
	1978	1973	1978	1972/80
Orbit Launch Mass (lb):	1,013,920	1,278,665	617,760	696,491

ORBITAL LAUNCH VEHICLE

A modified S-II stage was investigated as the orbital launch vehicle for the Mars and Venus flyby missions. There are two basic possibilities of using the S-II stage for launch from Earth orbit, as discussed below.

Reuse of the S-II Stage

From Earth launch, the S-II and the spacecraft reach Earth orbit toegther. Thereafter, the S-II stage is refilled with both LOX and liquid H_2, and

reused for orbital launch. The mass of a S-II stage at cutoff (without reserves) is assumed to be 96,560 pounds (Nov. 1964: 92,200 pounds). To this must be added:

LOX tanker docking, LOX transfer LH_2 tanker docking, LH_2 transfer Insulation/Meteoroid Protection Restart capability	13,000 lb. (assumed)

This addition gives a total stage cutoff weight of 109,560 pounds.

The aft skirt assembly of the spacecraft is now just an interstage (plus IU),* reducing its mass from 24,576 pounds to 11,576 pounds, resulting in a mass savings of 13,000 pounds. The maximum propellant loading of the stage is 930,000 pounds; the mixture ratio is 5:1; the I_{sp} is 425 seconds. The minimum propellant loading requirement for Mars occurs in 1978 with 703,000 pounds; the maximum in 1973 with 928,000 pounds. It was assumed that the full tankage size is used for all missions. With tailored tankage, the following mass savings can be obtained:

$$\text{Mass saving (lb)} = 1{,}516 \cdot \frac{930{,}000 - \text{prop. loading}}{100{,}000}$$

This result in a maximum saving (for 1978) of 3,441 pounds of inert S-II stage mass. This was, to repeat, not utilized in this study for Mars missions.

Special S-II Launch Vehicle: S-IIB

Mars

The S-IIB stage is the case primarily investigated in this study; the following major changes have been assumed for the S-II stage:

(1) Increased insulation for the liquid hydrogen, which is orbited with the stage, thus eliminating LH_2 tankers and associated equipment.

(2) The number of J-2 engines is reduced from five to three, resulting in a mass savings of 6,600 pounds.

(3) LOX tanker, docking and transfer provisions have to be added.

(4) The spacecraft docking provisions have to be added but are charged to aft structure (the addition amounts to 5,700 pounds).

It is assumed that the changes in items (1) through (3) leave the cutoff mass of 96,560 pounds; this does not include the 5,700-pound addition to the aft structure, which is carried with the S-IIB.

Venus

First, the same stage used for Mars was used for Venus. However, as an alternative, a special Venus injection stage was visualized. The outstanding features of this special stage are:

(1) The number of engines was reduced to two.

(2) The tankage was tailored for a much smaller capacity. This was due to the propellant loading being so far below 930,000 pounds (only 394,000 pounds for 1978, and 461,000 pounds for 1972 and 1980). The mass of this stage is, in pounds:

$$93{,}230 - 1{,}516 \cdot \frac{930{,}000 - \text{propellant loading**}}{100{,}000}$$

The first number is equal to 96,560 pounds minus the mass of one J2 engine, or

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*Because no transfer of spacecraft to orbit launch vehicle is required.

**For the computations, 88,341 pounds has been used instead of 93,230 pounds; the difference of 4,889 pounds representing various improvements of the S-IIB stage. If these are not utilized, all orbital launch masses for Venus have to be increased by up to 14,000 pounds, about half of which will be LOX.

3,330 pounds. The second term represents the S-II stage mass saving due to reduced propellant capacity.

In the case of Venus, the mass at orbital launch is reduced significantly (for 1978, by 40,700 lb, or by 6.18%), whereas, in the Mars case, the potential reduction of orbital launch mass is less significant (for 1978, about 11,200 lb, or 1.11%). Therefore, for Venus this "short S-IIB" is recommended, for Mars the "standard S-IIB" is preferred.

Figures 3, 4, and 5 show the spacecraft dimensions, the longest and shortest orbital launch configuration, and the tallest Earth launch configuration, respectively.

ORBITAL OPERATIONS

Considerations of Mass Distribution Between Surface Launches

Typical sequence and payload mass distributions are:

1. Put unmanned spacecraft plus aft structure and rendezvous transfer propellants (in total, for Mars, up to 274,000 lb in 1973) into 185-km orbit.
2. Put a total of up to 772,864 pounds LOX (Mars, 1973) into 185 km orbit (4 LOX tankers, of 193,216 lb LOX capacity each).
3. Put S-IIB plus up to 154,573 pounds of liquid hydrogen (Mars, 1973) or a total mass of up to 256,833 pounds into the 485-km departure orbit.
4. Transfer crew into orbiting spacecraft.
5. Transfer spacecraft to 485 km orbit and dock to S-IIB. (Total speed required about 190 m/sec; with an average specific impulse of 397 sec, the required mass ratio is 1.05.)
6. Transfer LOX tankers to S-IIB and transfer to LOX to the S-IIB.

Total vehicle mass (Mars, 1973) in 485-km orbit:

Spacecraft:	248,968
LOX:	772,864
S-IIB/LH_2:	256,833
TOTAL	1,278,665

7. Launch from 485-km orbit into interplanetary trajectory. The lightest Mars missions occurs for 1978:

Spacecraft in 485-km orbit:	208,805 lb
3 LOX tankers, each 188,571 lb	565,712 lb
S-IIB/117, 143 lb LH_2/20,000 lb LOX:	239,403 lb
Launch mass in 485-km orbit:	1,013,920 lb

Therefore, we need for Mars either five or six successful Saturn V launches. For Venus and the tailored S-IIB stage the following cases are extreme:

	1972/80	1978
Spacecraft in 485-km orbit	173,658	162,583
Two LOX tankers each		
141,783 LOX	283,565	
109,014 LOX		218,027
S-IIB		
72,713 lb LH_2/80,000 lb LOX	239,268	
61,605 lb LH_2/90,000 lb LOX		237,150
Launch mass in 485-km orbit	696,491	617,760

So in this case, four successful Saturn V launches are required, and the tanker capacity demand is quite modest. Indeed, it is not impossible that the 1978 Venus case could be performed by only three successful Saturn V flights, if, e.g., the tanker capacity could be increased to 198,500 pounds, and the S-IIB could be orbited with 61,605 pounds of LH_2 and 100,527 pounds of LOX, to give a total of 256,677 pounds, which is about equal to the maximum 1973 Mars requirement.

A remark as to LOX or LH_2 tankers based upon Saturn V: It has been assumed that the tanker is a self-contained stage put, by Saturn V, into the 185-km orbit. Further maneuvering is done by the tanker.

The following data are obtained from NASA Contract NAS8-11326, Orbital Tanker Design Study, Lockheed Aircraft Company.

Vehicle Assumed in NAS8-11326	185-km Orbit Delivered Payload (lb)	LOX Tanker Useful LOX Delivered to 485 km (lb)	LH_2 Tanker Useful LH_2, Delivered to 485 km (lb)
Saturn V	242,000	195,000	170,000
Saturn V (improved)	329,000	270,000	233,000
Assumed in this study		by scaling	
Saturn V (improved)	290,000	238,000	not required

Since the above study appears to be quite optimistic in some assumptions, only 190,000 pounds of useful LOX have been utilized.

Consideration of Facilities and Launch Schedule

The longest of the vehicles is the spacecraft launcher; it measures 332 feet from the bottom of the S-IC stage to the top of the spacecraft shroud. This maximum height can be handled in the present facilities for Complex 39.

So far, only gross masses have been listed. The problem of optimizing the scheduling of payloads into orbit and the sequence of orbital launch operations are quite complicated. As an example for a Mars flyby mission occurring in September, 1975, the following Saturn V launches would be required: (One backup for each launch configuration is assumed.)

Launched Configuration	No. of Launches (including backup)
Spacecraft	2
S-IIB	2
LOX Tankers	5
TOTAL	9

The sequence of events for the Earth orbital launch operations are shown in Figure 6. The prime constraint in the development of this sequence was to minimize the orbital staytime of the S-IIB stage since it is launched (from Earth) with the liquid hydrogen aboard. The backup launches are considered to be in the pad and available to be launched within 24 hours. Asseumptions for the development of this sequence are that some type of facility exists in Earth orbit which will house nine checkout crew personnel together with the appropriate checkout and maintenance equipment and space parts. The mission crew is also assumed to be housed within this facility until they board the mission spacecraft at T-20 hours. It appears that the orbital staytime for the S-IIB stage can be minimized at approximately 50 hours. Based on rough heat transfer calculations (insulation, sub-cooling, and shadow shield), it appears that the S-IIB stage can store the liquid hydrogen in Earth orbit under nonvented conditions for around 72 hours. Other schemes for increasing this time could be investigated. Perhaps a blanket of insulation would be practical if it was considered as a harness consisting of a wire structure covered with super insulation into which the S-IIB was placed. The sequence of ground launches to support the requirements of Figure 6 are in the reverse order of the numbering system shown in the figure. In other words, No. 9 is

launched first; No. 8 second, etc. Launches No. 9 and 8 were constrained to September 15, 1975, and September 13, 1975, respectively; and the remainder are launched as a function of Complex 39 capability.

A summary chart of the Earth launch sequence and launch schedules is shown in Figure 7. Care should be taken to note that the resulting firing dates are a direct function of the listed assumptions. The listed assumptions as far as facilities are concerned would incur a minimum cost to the program since most are presently planned. (Equipment for one bay, one arming tower, and two LUT (Launch Umbilical Tower) refurbish areas were added above and beyond presently planned facilities). The reduced times for the major operations appear to be logical at this time.

MISSION INTEGRATION

Launch from Orbit

Let us assume that engine ignition of the orbit launch vehicle (S-IIB) has occurred successfully. The powered trajectory is shown in Figures 8 and 9 (for Mars, 1975).

Injection occurs in the example shown 7 minutes and 57.87 seconds after ignition. A vernier phase will follow establishing the injection condition as accurately as possible. Corrections will be based on the best information available: inertial, optical/celestial, and/or tracking from Earth. The figures show that the powered trajectory has no unusual features. It is worth noting that only during the final 158 seconds (i.e., during the final 33% of the powered trajectory time) is the hyperbolic excess speed real. Note the relatively short thrusting time in Figure 9.

Orbit Launch Window

There are three launch windows interacting which have to be considered. They are as follows:

1. Interplanetary Window

By just looking at the end points and trajectory constraints (in this case: Earth launch -- flyby encounter with planet -- back to Earth), we can see that there is a best relative astronomical constellation between the celestial bodies involved (Earth, planet) so that injection speed requirement is a minimum. To launch at some time other than the optimum leads to an increased speed requirement.

2. Regression of the Nodes

If we launch from a satellite orbit, the speed increment should be tangential to the satellite orbit. If both the initial satellite orbit and the required Earth-referenced departure velocity (magnitude and direction) are given, then generally it will not be possible to connect the two with a single tangential impulse. We must either connect them with multiple impulses (excluded from this study because of complexity of re-ignition), or we must connect them with a non-tangential impulse. This again leads to an increased speed requirement.

We shall select our assembly orbit such that, for the ideal launch time for the planetary window, tangential launch is possible. This means that if launch really occurs on time, the tangential condition will also be fulfilled.

Unfortunately, this tangential condition cannot be fulfilled permanently for two reasons. The minor one is that, as time changes, changes occur in both direction and magnitude of the initial speed to go from where you are (Earth) to where you want to go (Mars), because of the relative motion of the endpoints. The second and major reason is due to the regression of the nodes (typically, 6.8 degrees/day), a consequence of the motion of the axis of the satellite plane around the Earth's axis mainly due to the Earth oblateness.

Any launch off the optimum time will, therefore, generally lead to a non-tangential thrusting requirement, which in turn leads to an increased speed requirement on part of the orbit launch vehicle ("Dog-leg loss").

3. Push-Button Window

The true anomaly of the orbiting space vehicle goes, during one revolution, through 2 π radians. Thus, the instantaneous velocity vector also goes through 2 π radians. In the ideal co-planar case, there will be one instant when the instantaneous velocity and the velocity increment to be added by propulsion are exactly parallel: this is the ideal time of launch.

Any deviation from this time leads to a non-parallel thrusting requirement, thus to an increased speed requirement.

Figure 10 depicts a typical orbital launch window situation. It should be mentioned that the nodal regression windows can show large deviations from the "average" behavior, depending upon the inclination of the orbit plane and the declination of the asymptotic departure velocity. Of course, in order to have the "plane window curve" touch the planetary window curve, the inclination has to be larger than the declination; this is fulfilled, see GD/Fort Worth Report FZA-391.

Since the actual computation is quite complicated, a simplified method was used:

(1) Compute the optimum case, having 2 percent performance reserve in braking stage.

(2) Just with regard to the planetary window, design the vehicle so that it can perform any mission $\pm$ 14 days around the optimum.

(3) Increase the injection speed capability by 100 meters/sec, for plane window and push-button window.

(4) Add 2 percent injection performance reserve.

It will be determined in the next phase of this study, for a typical case, what actual launch window can be obtained using this procedure. Tentatively, the situation looks as follows: A push-button error of $\pm$ 1 minute will require a speed penalty of, typically, only 25 m/sec; the remaining 75 m/sec (of the assumed 100 m/sec) is available for the window due to regression of the plane. It appears that this, typically, will result in a window of 5 to 10 days duration (or, possibly of operational advantage, two smaller windows within the 28-day interplanetary window).

These tentative conclusions are based upon report IN-P&VE-A-64-7, May 14, 1964, "Earth Orbital Launch Windows for Mars Missions," R. M. Croft, and some additional information made available by Mr. W. R. Perry, P&VE, Marshall Space Flight Center.

Free Fall to Target (130 Days)

Some major points of interest in this phase of the mission are:
Zero gravity
Attitude control available (propellant: 2.5 lb/day)
Astrionics: Inertial/Celestial; crew helping (e.g., measurement of photographs), with possibly help from Earth (computation or tracking).
Midcourse corrections: I_{sp} = 313 sec; LEM descent engine of 10,000 pounds thrust.

ΔV_1 = 150 m/sec, for Earth-target planet correction
ΔV_2 = 150 m/sec, during encounter
ΔV_3 = 200 m/sec, for target planet - Earth correction and Earth braking phase positioning

Encounter Phase

At encounter with the target planet, 10,000 pounds of probes will be released. These probes, and the method of releasing them, will need to be determined at a later date. The probes could be designed to orbit, float in the atmosphere, land or possibly perform aerodynamic braking tests at the target planet. The passing spacecraft would monitor, store, partially evaluate and then relay the data back to Earth. This concept would be combining the best advantages of the unmanned probes with the best advantages of having man in the mission. The probes may later, at reduced bandwidth, communicate directly with Earth. The encounter with Mars will be hyperbolic with the pericenter of the hyperbolla being over the dark side near the terminator. The encounter speeds are high (9.5 - 12 km/sec at pericenter), the staytime near Mars short (typically, one hour within 18,520 km of the center of Mars, or about 150 minutes within double this distance).

These short encounter durations are misleading, since even with only a modest telescope on board the vehicle, the target surface resolution is better than from Earth with the largest practical telescope during a good opposition, for more than a month and many observations depend not so much upon resolution as upon the avoidance of looking through the Earth's atmosphere: in this respect, conditions are favorable for more than 100 days.

Free Fall Back to Earth (539 Days)

This part of the mission is similar to paragraph 3 except for data transmission and transfer of records into the command module. Approximately 50 days before Earth rendezvous, the crew begins checkout, repair, etc., of the command module, service module, and hangar release mechanism.

Earth Landing

Approximately six days before Earth entry, the crew enters the command module. This will allow about four days of "trial life" in the module. At the end of these four days the command/service modules are separated from the spacecraft. Towards the end of the following two days, the service module is ignited to provide retro-thrust, slowing the command module to Apollo entry conditions. From there the touchdown occurs as it does in the Apollo lunar mission.

Guidance for the entry maneuver is provided by an inertial/celestial system, or an inertial system, and/or radio from Earth.

Trajectory Information

A high precision trajectory computational program has been developed. This proved that the Lockheed approximate program (originated under NASA Contract NAS8-5024) is accurate enough for preliminary work, with the possible exception of guidance sensitivity effects.

Figures 11 through 14 are four typical results, where all motions are projected into the plane of the ecliptic. Since inclinations are small (Mars: 1°51'; Mars flyby trajectory: outbound, 1°37' - inbound, 5°36'; Venus: 3°24'; Venusian flyby trajectory: outbound, 1°39' - inbound 1°28'), this is a realistic presentation.

Crew Size

A crew size of three was picked, not only because of the compatibility with the present Apollo command module, but also because this size appears to be optimum from a task analysis/efficiency point of view.

The following quote is taken from the General Dynamics/Fort Worth publication MRO-89, "Mars Flyby Crew Operations and Crew Requirements."

"With respect to unscheduled maintenance operations the probability of a two-man crew holding schedule is .90 under two conditions. Either the basic spacecraft operations are limited to 16 hours per day, or continuous basic spacecraft operations are allowed and the maintenance efficiency is significantly increased.

"The probability of a three-man crew holding schedule is .995 assuming the lowest anticipated maintenance efficiency level, and greater than .995 for increasing efficiency levels.

"The requirements of presently proposed sensors and essential spacecraft operations during the flyby phase indicates that a crew of two men has a marginal capability of carrying out a minimum of sensor requirements, whereas a crew of three is capable of accomplishing the necessary sensor operations. It was also determined that a crew of four is not required.

Emergency Situations

If an emergency occurs in Earth orbit, normal orbital operations procedures will be followed. If, however, an emergency occurs in the boost phase there are two alternatives. The spacecraft can remain in an elliptical orbit and await rescue from Earth, or the Apollo service module could be ignited to either establish a desired orbit or to return directly to Earth, depending upon the situation.

After injection, the spacecraft is committed to the mission. However, there is the possibility of on-board repair, or, in case of loss of a crew member, it would be possible for a two-man crew to complete the mission.

Much more work needs to be done in this area.

Radiation Protection

The time period considered for the Mars mission is September 1975 to July

1977. This is a near minimum period of solar activity. During the 22-month period only one major flare should be expected. With the shielding provided by an Apollo equivalent space ship (6 gm/cm^2) the total skin dose for the total trip should not exceed 100 rads with a 99+% safety factor. Thus, the probability is less than .01 that a greater total dose would be obtained. The bone marrow dose would be only 30 rads for the total trip. The galactic cosmic rays would provide a dose from 20 to 30 rads for the entire trip but they cannot be shielded against because of their extreme energy. Thus with such a small risk factor it seems that a flare shelter is not needed during the 1975 to 1977 period. However, the on-board supplies should be distributed in such a manner as to provide additional protection during a major flare event (12 to 48 hours).

The above comments are true for the Venus flight in 1973 but the hazard is even smaller. During this period there should be no major flare events. However, the ship is approaching the Sun and the smaller flares may be more important, but the radiation risk will still be probably less than the Mars flight in 1975 to 1977.

SCIENTIFIC PAYLOAD

As mentioned previously, there are 10,000 pounds of scientific probes to be dropped during the encounter, and 1,000 pounds of scientific payload which will remain with the flyby spacecraft. Following is a listing of possible information to be gathered:

Interplanetary

1. Plasma and magnetic effects upon passage through Earth's Geomagnetic Tail.
2. Studies of the Solar Spectrum-Radio through X-Ray Region.
3. Observation of solar activity on Sun's back side for correlation with observations on the Earth side.
4. Local solar magnetic and electric fields.
5. Meteoroid - cosmic dust fluxes (with the exception of extremely minute particles, these fluxes may be vanishingly small).
6. Neutron spectrometry.
7. Structure of solar plasma (wind) streams, globes or shells. Composition and vector velocity - flux distributions and temporal variations.
8. Studies of cosmic rays - composition, velocity - flux distributions and temporal variations.
9. Relativity and gravitational experiments.
10. Biological experiments - probably in main an extrapolation of then available MOL data.

Planetary

1. Presence and structure of radiation belts.
2. Presence of a hydrogen "geo" corona.
3. Planetary emissions-radio, infrared, visible, ultra violet.
4. To distinguish atmospheric emissions and aurorae (Venus).
5. Atmospheric composition and physical parameter profiles.
6. Atmospheric refraction effects.
7. Surface properties.
8. Venus' rotation period and axial inclination.
9. The ashen light or dark side luminosity of Venus.
10. Venus aurorae.
11. Magnetic field pattern and strength.
12. Deimos and Phobos, their sizes and other properties.
13. Dark areas of Mars and their change with time.
14. The yellow clouds of Mars.
15. The blue haze of Mars.

It is felt, however, that the major emphasis of the manned flyby-unmanned probe combination should be focused on assisting the later landing missions.

A large savings in weight can be realized in landing or orbiting missions if aerodynamic braking is used at the planet. This method cannot be used, however, until the atmospheric properties are known, and aerodynamic braking designs (both landers and skippers) have been tested.

This information could be obtained by including in the probes various types of investigation, such as landers, atmospheric floaters, skippers, orbiters, and possibly probes which will be specialized to perform aerodynamic entry tests as to both designs and materials.

If the probes determine that there are usable indigenous materials on Mars, such as water, then the first landing mission could realize a savings in weight and complexity of the mission.

MISSION SCHEDULE AND COST ANALYSIS

The purpose of the following paragraphs is to summarize the major results of the schedule and cost analysis for the Venus and Mars flyby missions. The discussion will present the major assumptions and methodology upon which the costing and scheduling was based, the schedules themselves on the basis of both launch and design and development events, the design and development and operational costs, and the manner in which these costs are translated into obligational funding requirements.

In order to derive the schedules and costs underlying the Venus and Mars flyby missions, certain major assumptions were made. First, the mission attempt dates were established to be 1975 and 1978, respectively, for the Venus and Mars missions, and it was decided that only one mission attempt each would be made.

The launch vehicle to be used for each of the missions was assumed to be an improved Saturn V configuration known as MS-V-1. This vehicle is currently being studied in the Saturn V improvement effort, which is being conducted both inhouse and under contract. The major changes involved are uprating of the F-1 engine to 1.8 million pounds of thrust and making corresponding tankage changes in both the first and second stages. The additional payload capability of this configuration will allow the desired payloads for the Venus and Mars flyby missions in the 485-km orbit to be achieved. It was assumed that no charge for the design and development of this configuration is absorbed by the flyby missions.

All of the mission hardware used in both flyby missions was costed against the background of a total space program involving many other projects using this or similar hardware. This was done to account for the commonality aspects. In order to derive this background of other usage for the hardware involved, certain mission mixes making up a total space program were used, based on previous work performed by the Future Projects Office of MSFC. The effect of this total Post-Apollo space program can be summarized by indicating the utility of the Saturn IB, Saturn V, and Apollo command and service modules in the years 1968 through 1978. The Saturn IB vehicle, after flight 212, was assumed to have an average launch rate of six per year through 1978. The Saturn V launch vehicle, after flight 515, was assumed to have a 6 per year launch rate through 1971, 9 flights in 1972, and 12 flights per year from 1973 through 1978. It was assumed that approximately two-thirds of all of the Saturn IB and Saturn V flights will have an Apollo command and service module as the payload. With these assumptions it was possible to use cost improvement curves on the Saturn vehicles and the Apollo command and service modules.

The design and development cost of the LOX Tanker and S-IIB Orbital Launch Vehicle, are fully charged against the Mars and Venus flyby missions. This assumption was arbitrary, since it was not known how much other orbital activity might utilize these two vehicles. Some of these design and development charges for these two vehicles would be charged against such alternate projects, thus decreasing the cost of the flyby missions. However, in using these vehicles, orbital operations are involved, and the design and development costs for developing this orbital operations capability were not charged against the flyby missions. These two factors tend to offset each other somewhat.

It was assumed that no funds would be available for new projects starts until Fiscal Year 1969. This major constraint was used for all funding and scheduling purposes in design and development which must take place in order to accomplish the flyby missions.

The design, development, test, and launch schedule is shown in Figure 15 for the major elements involved. The program definition would begin in mid-1968 and be 12 months in duration. The systems integration and project management would begin at the same time and last through the duration of the missions.

Three of the major developments which must take place are the spacecraft itself, the astrionics required for the mission, and the LOX tanker. All three of these developments should begin by the last quarter of 1968. The design, development, and test of the spacecraft and astrionics for the mission would run through the 1975 Venus flyby date. The LOX tanker development would be concluded in 1974 prior to the Venus flyby attempt. During the development of the LOX tanker, two flight tests of this item would be carried out using Saturn V launch vehicles in late 1973 and in mid-1974. One test unit of the spacecraft itself would be orbited by a Saturn V in 1974 for purposes of development testing and crew familarization and training. A complete set of astrionics hardware would be orbited by a Saturn IB in 1974 for development testing and observation under long lifetime space environment conditions.

The design, development, and test for the S-IIB Orbit Launch Vehicle, was assumed to be a major modification program to be basic S-II stage. These modifications start in late 1969 and terminate in 1974. Flight tests of the S-IIB would be carried out in 1973 and 1974 using Saturn V launch vehicles.

Modifications to the Apollo service module would begin in mid-1970 and extend through 1974. This development and design work would be necessary to extend the burning time duration of the service module, which is required for the Mars flyby missions. It is also to check and test the lifetime duration of this module in the space environment. It was assumed that no special launch vehicle would have to be set aside to test the modified Apollo service module, and the test would be carried out as a part of other missions.

The development of the aft skirt assembly, which is a propulsive orbital operations element, begins in late 1970 and runs through early 1975. A flight test unit of the aft skirt assembly would be flown in 1974 on the Saturn IB launch vehicle.

The development of the scientific probes and equipment for the Venus flyby would begin in mid-1970 and run through the third quarter of 1975. The corresponding development activity for the Mars flyby would begin in the last quarter of 1973 and run through 1977.

The flyby attempt for the Venus mission would be conducted in 1975 using four launches of the Saturn V vehicle. The first launch would be used to place the unmanned spacecraft into a 185-km orbit. The next two launches would place the tankers into the same orbit, and the last Saturn V launch would be used to place the S-IIB into a 485-km orbit. Each of these three launch phases is assumed to have one Saturn V launch vehicle as a spare, thus calling for a total of seven Saturn V vehicles for the Venus flyby mission. A single launch of the Saturn IB vehicle is required for crew transfer, and one Saturn IB space is used for this phase of the operational mode. The same sequence is required for the Mars flyby mission except that four tanker flights are required. The spares requirement for both Saturn V and Saturn IB's are assumed to be the same as in the Venus mission.

The design, development, and operational costs are shown in Figure 16. The distinction between design and development costs and operational costs is an arbitrary one, chosen mainly for the convenience of illustrating the additional cost, which would be required for repeated mission attempts. No attempt was made to prorate the design and development costs between the Venus and Mars missions, except in the case of the scientific probes and equipment. Thus, the design and development charges in the main are for both the Venus and Mars missions.

The total development cost for the modification of the S-II stage into an orbital launch vehicle is 425 million dollars. This figure includes modifications to the stage in the form of removing two of the J-2 engines, providing orbital docking, providing checkout and launch hardware and procedures, and providing attitude control features. The cost also includes provisions for the launch on Saturn V vehicles of two flight test units.

The design and development of the LOX tanker was assumed to be a new development, costing 380 million dollars. This estimate was based on the cost estimating relationships for launch vehicle stages currently in use in the Future Projects Office's Launch Vehicle Cost Model. It includes provisions for the test of two flight test items using Saturn V launch vehicles. The design and development of the aft skirt assembly was also costed using the cost estimating relationships for launch vehicle stages with some special provisions made for its docking features. The aft skirt assembly design and development cost of 165 million dollars includes the launch of a flight test unit on a Saturn IB launch vehicle.

The Apollo service module requires modifications to allow it to be used in the flyby missions as stated earlier. These modifications were assumed to be 10 percent of the basic design and development cost for the Apollo service module and includes provisions for buying one item of flight test hardware. The total cost for these modifications to the Apollo service module is 115 million dollars.

The design and development of the astrionics equipment for the Mars and Venus flyby missions was assumed to be a problem of equal magnitude to the lunar mission astrionics problem. Special emphasis and consideration must be placed on the long lifetime required for these astrionics elements and also on the communications requirements. The cost shown for the astrionics equipment, 325 million dollars, includes a test flight of this equipment on the Saturn IB vehicle.

The spacecraft design and development cost, 1,563 million dollars, was based on cost estimating relationships derived using Apollo, Gemini, and Mercury spacecraft cost data. This cost estimate includes the purchase of one flight test unit and its launch on a Saturn V.

An estimate of the design and development charges for the scientific probes was made using a cost per pound figure of 20,000 dollars. There are 10,000 pounds of scientific probes on each flyby mission and 1,000 pounds of scientific equipment within the spacecraft itself. This results in a design and development cost for each mission of 220 million dollars. Much of the hardware between the Venus and the Mars missions would be similar, but it is felt that the time delay between the two missions and the knowledge gained from conducting the Venus mission would essentially cause new development and design of the Mars scientific probes and equipment.

The systems integration and project management for the Mars and Venus flyby is assumed to be 10 percent of the total design and development cost incurred, which is based on past program experience. This cost, 340 million dollars, is funded uniformly over the entire mission time spectrum Fiscal Year 1969 through Fiscal Year 1978.

The operational costs for the 1975 Venus mission and the 1978 Mars mission are also shown in Figure 16. The Saturn V launch vehicle configuration used for these missions is composed of the S-IC, the S-II, and the instrument unit. A 90 percent learning curve for the Saturn V was used and the learning curve was entered at 62 prior units. This yielded an average unit cost for the Saturn V of 70 million dollars. The Saturn IB vehicle also uses a 90 percent learning curve, which is entered at 52 prior units to yield an average unit cost for the 1975 Venus mission of 22 million dollars. The Saturn V average unit cost for the 1978 Mars mission assumes 98 prior units, yielding an average unit cost of 65 million dollars. The Saturn IB for the Mars mission assumes 70 prior units yielding an average unit cost of 20 million dollars. The hardware cost for the Apollo command and service module is calculated by using a 95 percent learning curve which is entered at 70 prior units for the 1975 Venus mission and 100 prior units for the 1978 Mars mission. This yields an average unit cost of 72 million dollars and 69 million dollars for the Venus and Mars missions, respectively. The hardware cost for the Saturn V, the Saturn IB, and the Apollo command and service module was based on the best currently available program estimates for these vehicles.

The S-IIB, aft skirt assembly, LOX tanker, and spacecraft, because of their complexity and small number of units involved, are assumed to have no significant improvement slope. The hardware cost for all of these elements, except the spacecraft, was derived using cost estimating relationships, based on the Future Projects Office's Launch Vehicle Cost Model. The spacecraft hardware cost was estimated using a cost estimating relationship, based upon Apollo, Gemini, and Mercury spacecraft data.

The total cost of the operational hardware involved in the Venus mission was estimated to be 1,279 million dollars, and the total cost for the Mars mission was estimated to be 1,392 million dollars. Although there is a large number of hardware units involved in the Mars mission, the cost would be only eight percent higher for this mission because of the learning curve. It is felt that a few additional Mars and Venus flyby missions would cost in the order of 1,200 million dollars each.

The detailed obligational funding requirements for the design and development of these items and also for the purchase of the operational hardware is shown in Figure 17. This obligational funding was calculated using beta curve

distributions, based on the schedules shown in Figure 15. Seven different beta curves were used, with the number of funding intervals varying between three and seven. The detailed funding requirements can be seen from the illustration. The peak funding for the design and development occurs in Fiscal Year 1972 and is 895 million dollars. The peak funding for the operational hardware occurs in 1974 and is 497 million dollars. The total funding peak occurs in 1973 and is 1,222 million dollars. The funding for the operational hardware broken down between the Mars and Venus missions can be seen in Figure 18. The cumulative rate of funding buildup required for both flyby missions can be seen in Figure 19.

MISSION WORTH ANALYSIS

In comparing various space program alternatives or different methods of accomplishing a given space program, the results in terms of yield and cost do not always provide sufficient information to make a proper or comprehensive comparison. For this reason an attempt has been made to derive a worth analysis methodology which will combine the various yield indices and costs with the stated objectives of any given space program. In other words, this worth analysis methodology attempts to answer the question of how well a particular space program satisfies the overall space program objectives, or which program plan of several seems to satisfy the objectives best. In this manner, the various yield indices, such as dollars per man trip to destination or dollars per pound to destination, can be combined into one common measure of worth. A more difficult problem is to establish a similar methodology or extend this methodology to the problem of measuring the relative worth of individual missions which are of interest. That is the problem of this analysis; to establish the worth of doing the manned Mars and Venus flyby missions. This problem will not be solved here; but a method of attack and some preliminary examples of types of results which can be obtained will be shown.

Several approaches should be taken in analyzing the worth of the Mars and Venus flyby missions. All of these alternative methods should be studied against the background of a total space program made up of many individual missions, as discussed earlier. In evaluating the relative worth of one program versus another, there are many different influences and interplays that must be considered, and it is necessary to analyze total programs in order to properly analyze individual missions within the programs, because of these interrelationships. This interplay exists in the form of the technical and mission feedbacks which exist between the various missions, prior developments of other missions which are applicable to the new missions of interest, the joint development of items of hardware which are applicable to more than one mission, the learning and production economics on hardware elements that are common to two or more missions, and the overall interplay which exists between various missions which are pursued to accomplish the stated objectives of the space program.

In the examples which are discussed below, the approach was to calculate the worth of a postulated total space program made up of many missions including both the Venus and Mars flyby missions in 1975 and 1978, respectively, with a follow-on landing mission for Mars. Against the worth which results from this basic background, three variations were made. The first of these was to conduct a program without either the Mars or Venus flyby missions or the Mars landing mission. The second variation removed the manned Mars missions while maintaining the manned Venus flyby mission, the third variation removed the manned Venus flyby mission while retaining the manned Mars missions. No accurate worth assessment of a Mars landing mission conducted without precursor manned flyby mission can be made without a detailed analysis. A very cursory analysis of the contribution, as far as dollar requirements are concerned, made by preceeding the manned Mars landing by the flyby missions has been made and will be briefly discussed below. Still another method of analyzing the worth of the flyby missions would be to attempt to construct a program which aimed at obtaining as closely as possible the same amount and quality of information with unmanned probes as could be obtained with the manned flybys and landing. This approach would also take a very detailed analysis which has not yet been conducted.

Figure 20 indicates the relative percentage reduction in total program worth, and in the planetary program worth, for the three alternatives to the basic space program. The general methodology used in deriving these preliminary estimates is discussed below. The basic space program with which

these three alternatives are compared allocated the total program worth between four basic areas in the following fashion:

1. Overall program indicators received 26 percent of the total worth points.
2. Orbital indicators received 40 percent of the total worth points.
3. Lunar indicators received 17 percent of the total worth points.
4. Planetary indicators received 17 percent of the total worth points.

From this allocation, it can be seen why the relative percentage reduction in the total program worth is relatively small when the manned planetary missions are removed. The construction of the measures which went into allocating the program worth between these four areas was, of course, based on value judgments. If these value judgments were changed then the percentage reduction in total program worth for these alternatives would change. For this reason, a more interesting consideration is the percentage reduction in the planetary program worth shown in the second line of Figure 20. The removal of the manned Venus flyby missions has little effect on the planetary program worth. This is because the indices which were used did not reflect as many indicators for the Venus mission as for the Mars mission, since it is not possible to land on Venus as it is on Mars. The numbers given in this example are preliminary and are only intended to illustrate what can be done to attack the problem of measuring the worth of the Venus and Mars flyby missions.

A preliminary estimate of the incremental cost necessary to perform the Mars landing mission having the Mars flyby missions as its precursor has been made. This estimate indicates that approximately two-thirds of the cost of the manned Mars flyby mission would be applicable to the manned Mars landing mission. This is given as a figure of merit only and is not intended to represent a detailed estimate. This estimate is probably on the high side and should be revised downward depending on the exact landing mode used. The figure is based on work performed by the Fugure Projects Office in analyzing several alternative space programs which included the Mars flyby and landing missions, and also work of a similar nature done under contracts to the Future Projects Office.

The methodology, which has been exercised in a preliminary fashion to derive the worth figures for the space program alternatives discussed above and which will be developed further, is rather simple in concept. The computation methodology can be represented by the matrix shown in Figure 21. The matrix consists of 20 objectives designated by the O_i's and up to 60 indices designed by the I_j's. These indices are partitioned into the four major groups discussed earlier and are designated by the brackets shown at the top of the figure. The objective weights, O_i, are inputs selected through subjective judgments or empirical decision rules.

The basic procedure then, is to compute the elements of this 20 by 60 matrix which are utilized, since each objective does not utilize every index. These elements can be summed to determine an individual objective worth, a group of objective worth, and total program worth. Each nonzero element in the matrix is a product of two terms. The a_{ij} term of each element is a function of the following two forms:

1. $e^{\frac{y-b}{\alpha}}$ $\qquad e^{\frac{y-b}{\alpha}}$

2. $1 - \frac{y-b}{\alpha}$ $\qquad 1 - e^{\frac{y-b}{\alpha}}$

Where y = the index value at the time of program evaluation, b = translation constant or reference year of index accomplishment and α = a parameter which is used to vary the slope of the worth function. The q_{ij} term of each nonzero element is a function of the weight that each index contributes to the program objectives. Various schemes can be used for computing these q_{ij} values. For the example given above, all indices used for each objective were assumed to have equal weight.

The methodology is structured such that either of the index functions can be used to compute the element of worth denoted by $a_{ij}\ q_{ij}$. The user can specify which index should be used for each objective and which function best relates

the worth of this index to the objective. Another desirable feature of this methodology is that it allows worth to be determined as a function of time which is particularly desirable and useful for determining the worth of the planetary flyby missions.

COMPLETE AERODYNAMIC BRAKING

The next two sections will give an outlook upon the magnitude of improvements, if certain technical capabilities were developed. In this section the reduction in orbital launch mass will be investigated, if complete aerodynamic braking of the command module was possible upon Earth return, without any rocket braking phase (Entry speeds: from Mars up to 4.8, from Venus up to 2.9 km/sec above parabolic).

Some tentative results are that the spacecraft is shorter, because the hangar can be shorter, and mass savings appear possible, as shown below for the nominal mission:

	Mars (lb)	Venus (lb)	Remarks
Nominal Spacecraft	189,929	150,331	
-10% Repressurization Gas:	-2,265	-1,505	Because of smaller hangar
-Service Module, Dry:	-10,894	-10,833	Not required
-Braking Propellant:	-47,790	-29,567	Not required
+Heat Shield Increase	+2,000	+1,500	Estimated
+Command Module, Support Structure	+ 500	+ 500	Estimated
+Increased Midcourse Propulsion	+10,000	+10,000	Higher precision for entry positioning
+Increased Midcourse Tankage/ Pressurization	+ 500	+ 500	Estimated
-Hangar Weight Reduction	-2,000	-2,000	Estimated
New Mass	139,980	118,926	
Mass Reduction	49,949	31,405	
Reduction in Percent of Initial Mass	26.3	20.9	

For the most demanding Mars year (1973), the initial mass in Earth orbit could be reduced from 1.28 million to about 0.97 million pounds, and for the easiest Mars year (1978) the reduction is from 1.01 million to about 0.85 million pounds: these results are quite impressive. In the case of Venus, the reduction of orbital launch mass is typically from 650,000 pounds to 580,000 pounds, a less significant amount. The reason is, of course, that Earth return speeds from the Venus mission are less than those from the Mars mission; therefore, less can be gained by substituting rocket retro-braking by aerodynamic braking.

NUCLEAR INJECTION FROM ORBIT

Although the study was concerned with all chemical propulsion, a limited analysis was performed on the use of solid core nuclear propulsion. For this investigation, the same computer program was utilized for the flight mechanics/ optimization procedure, and the same spacecraft was assumed. The only change consists of replacing the S-IIB stage by a nuclear injection stage. Figure 22 gives a description of the nuclear stage and spacecraft.

Two cases of nuclear capability were investigated. These are believed to represent reasonably optimistic and pessimistic limits of expected performance.

	Case 1	Case 2
Thrust (lb)	250,000	200,000
Specific Impulse (sec)	820	800
Stage Cutoff Mass (lb) (Including aft assembly)*	90,000+ +0.11(W - 130,000)	100,000+ +0.12(W - 130,000)

*The W refers to the liquid H_2 weight in pounds.

Following are some typical numbers for these two cases, versus a chemical S-IIB stage for a Venus and Mars mission.

Venus

Year = 1975 conjunction; trip-time = 368 days; no interplanetary window (This is done in order to get an optimistic lower limit.); 100 m/sec excess for all three launch windows; 2% ΔV flight performance reserve.

	Nuclear		Chemical
	Case 1	Case 2	
Orbit Launch Mass (lb)	388,290	414,246	634,230
Injection Propellant(lb)	148,600	162,800	381,620
Jettisoned after Injection (Incl. aft skirt) (lb)	91,683	103,439	104,603
Injected Payload (lb)	148,007	148,007	148,007

Mars

Year = 1975; trip-time = 682 days; 28 day interplanetary window; 100 m/sec excess for plane/push-button windows; 2% ΔV flight performance reserves.

	Nuclear		Chemical
	Case 1	Case 2	
Orbit Launch Mass (lb)	578,241	626,164	1,098,376
Injection Propellant (lb)	278,000	310,600	782,761
Jettisoned after Injection (incl. aft skirt) (lb)	105,762	121,085	121,136
Injected Payload (lb)	194,479	194,479	194,479

In both the Venus and Mars cases, the nuclear orbital mass requirement is only $\sim$ 59% of that required for the chemical case -- a significant reduction. Again, the Mars case profits more (reduction to 55%) than the Venus case (63%)

The effects of nuclear versus chemical propulsion on cost and schedule remain to be investigated.

As usual for nuclear vehicles, the mass reduction goes hand in hand with a length increase of the orbit launch vehicle. The total orbital launch vehicle can be about 50 percent longer than its chemical counterpart.

Figure 23 shows the launch configuration of the nuclear stage.

An attractive program might result from going to Venus in 1975 chemically (about 650,000 pounds in Earth orbit), and going with the same spacecraft in 1978 to Mars with nuclear propulsion, requiring about 600,000 pounds mass in Earth orbit. Thus, the Earth surface to orbit transport task would be very similar for both missions.

A breakdown of the launch to orbit and orbital operations requirements is as follows:

First launch: Spacecraft plus aft end, etc.
Second launch: Orbital launcher

	Heaviest Case (Mars) (lb)	Lightest Case (Venus) (lb)
Stage	100,000	70,000
LH_2	150,000	150,000
TOTAL	250,000	220,000
LH_2 to be tanked:	160,000	None

Third launch: One tanker for the Mars case only. (For orbital staytime reasons, the tanker would probably be the second launch.)

For the nuclear case, we need: Venus, two launches, no LH_2 tanker. Mars, three launches, LH_2 tanker required.

In the chemical case, we required LOX tanking only. Here we need the LH_2 tanker only. Since it appears unrealistic to assume two tanker developments, a decision should be made early as to whether preparations are started for chemical or for nuclear operations.*

If we assume the need for one spare vehicle of each type, we must prepare for:

Venus: 4 Saturn V launches
Mars: 6 Saturn V launches

In the chemical case, we had up to 7 and 9 vehicles instead.

The Saturn V Earth launch vehicle requirements are summarized below.

Earth Braking	Orbit Launch			
	Chemical		Nuclear	
	Mars	Venus	Mars	Venus
Retro	6	4	3	2
Aero	5	3	2	2

Saturn V Launch Vehicle Requirements For
Mars/Venus Flyby missions (Without Spares)

Earth Braking	Orbit Launch			
	Chemical		Nuclear	
	Mars	Venus	Mars	Venus
Retro	9	7	6	4
Aero	8	6	4	4

Saturn V. Launch Vehicle Requirements For
Mars/Venus Flyby Missions (With Spares)

PROBLEM AREAS TO BE INVESTIGATED

Following is a listing of problem areas to be investigated.

1. Spacecraft
   a. Development of internal power profile.
   b. Conceptual design of tailored power supply system, and assessment of radioisotope availability.
   c. Further analysis of life support system.
   d. Refined analysis of the attitude control requirements.
   e. Conceptual design of astrionics systems and procedures.
   f. Conceptual design of data storage and communication systems.
   g. Sensitivity analysis: effects od e.g., ± one crew member; ±1000 lb scientific equipment, ±3 lb daily leakage rate, etc.
   h. Probe launcher design.
   i. Determination and effects of realistic S-IIB data.
   j. Design of command module/service module and hangar release mechanism.
2. Scientific Payload
   A Conceptual definition of this is needed.
3. Earth Landing
   a. Investigation of practicality to use service module for braking, and alternate solution, if required.

Possibly, no tanking of LH_2 should be performed at all, but adding of complete LH_2 tanks, which top the main tank. Tank staging could lead to a small further improvement.

b. Investigation of gains due to "tailor-made service module."
c. Parametric investigation of mass reduction due to higher (above parabolic) allowable entry speeds for aerodynamic entry.
d. Landing point control (footprint) analysis.

4. Orbital Operations.
   a. Second look at orbital operations which were assumed for phase I, with detailed LH_2 storage analysis.
   b. Investigate possibilities to increase permissible orbital LH_2 staytime.
   c. Comparison (including economy, reliability, etc.) of S-IIB (as per phase I) versus S-II reuse for orbital launch (+ LH_2 tanker).
   d. Comparison of S-IIB (as per phase I) versus tendem staged S-IVB vehicles.
   e. Second look -- including the orbital operations procedures -- of using a nuclear orbit launcher.
5. Schedule/Cost/Mission Worth Analysis
   a. Refinements of the phase I effort in this area.
   b. Inclusion of new elements (e.g., S-IVB as an orbit launch vehicle) from Phase II.
   c. Cost analysis of nuclear and mixed programs.
   d. Pertinent results from the National Program Simulation Model.
6. Flight Mechanics
   a. Launch window situation and utilization of available space stations.
   b. An analysis of midcourse propulsion requirements, using the precision flight mechanics program.
   c. Investigation of possible advantages resulting from use of propulsion during the planetary encounter or other phases.
7. Mission Analysis
   Emergency situations and procedures.

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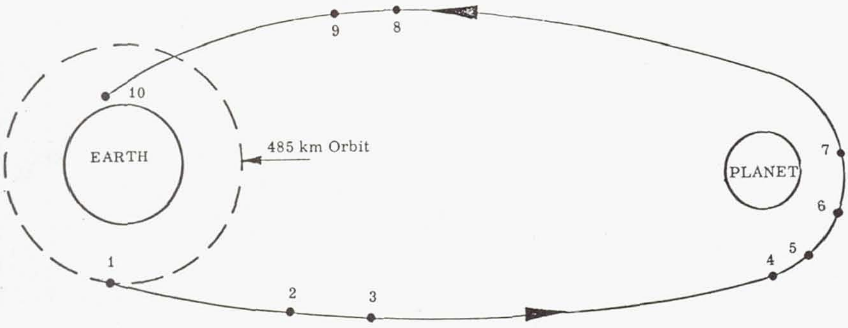


1. Spacecraft Injected into Interplanetary Trajectory from 485 km Orbit.
2. Jettison Consumed Life Support.
3. First Midcourse Correction (150 m/sec).
4. Jettison Consumed Life Support.
5. Jettison 5,000 pound Scientific Probe.
6. Second Midcourse Correction, (150 m/sec).
7. Jettison 5,000 pound Scientific Probe.
8. Jettison Consumed Life Support.
9. Third Midcourse Correction (200 m/sec).
10. Earth Braking and Atmosphere Reentry (11,000 m/sec at 120 km).

FIGURE 1

Typical Mission Profile

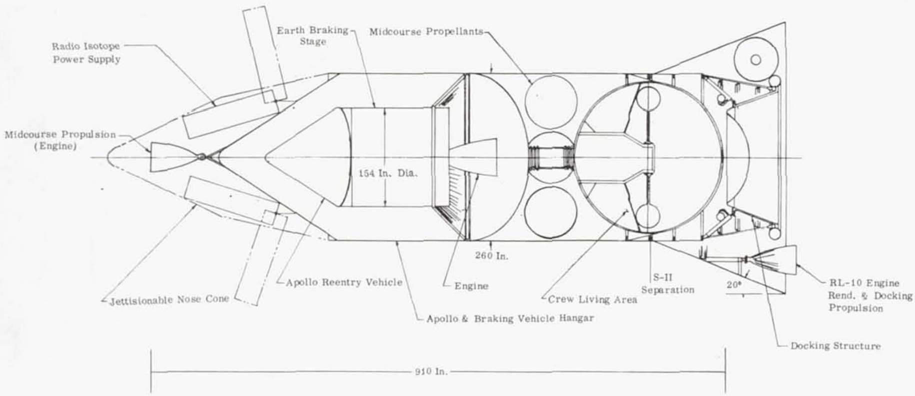


FIGURE 2

Flyby Spacecraft

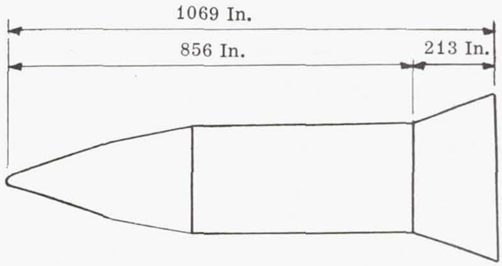


FIGURE 3

Spacecraft and Aft End Dimensions

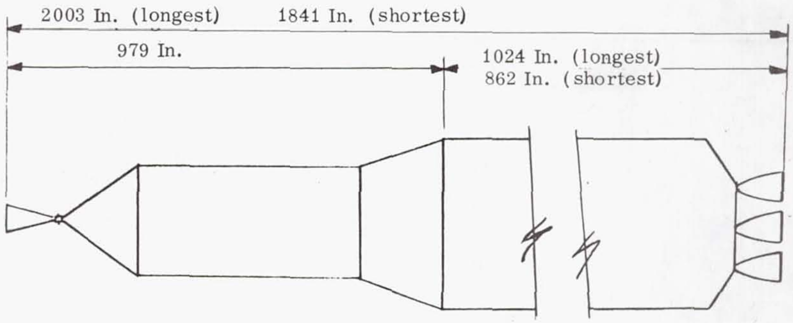


FIGURE 4

Longest and Shortest Orbital Launch Configuration

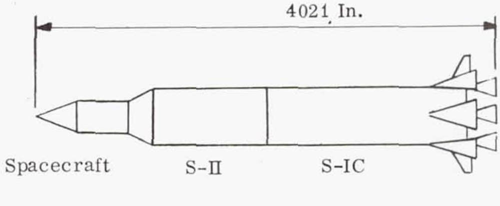


FIGURE 5

Tallest Earth-To-Orbit Configuration

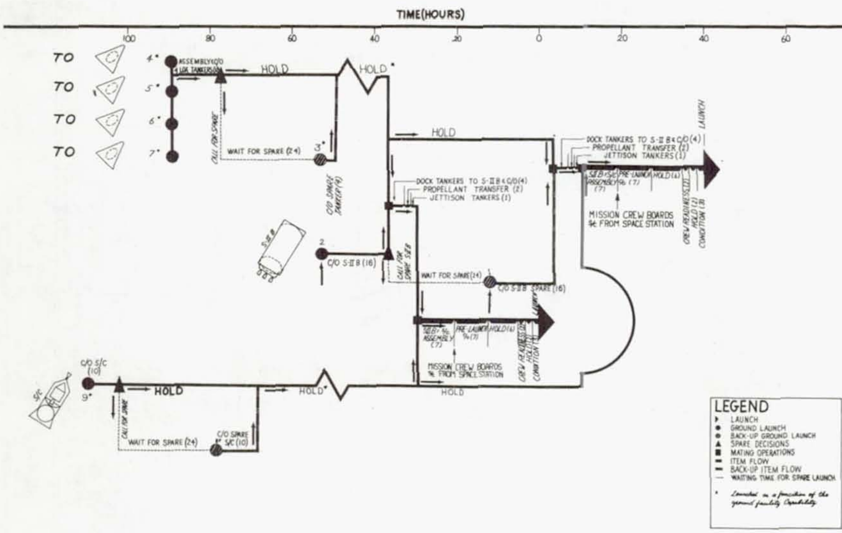


FIGURE 6

Orbit Launch Operations and Sequence (Mars Flyby Mission)

ASSUMPTIONS

- 4 LUTS
- 4 HIGH BAYS
- 4 FIRING ROOMS
- 2 PADS
- 2 ARMING TOWERS
- 4 LUT REFURBISH AREAS
- CRAWLERS NOT CONSIDERED
- THAT IN THE 1975 TIME PERIOD THE OPERATIONS AT CAPE KENNEDY HAVE BEEN IMPROVED AND SIMPLIFIED TO THE EXTENT THAT THE FOLLOWING TIMES ARE APPLIABLE TO EACH MAJOR OPERATION
    - HIGH BAY OPERATIONS: 25 WORKING DAYS
    - PAD: 5 WORKING DAYS
    - LUT REFURBISH: 20 WORKING DAYS
- LAUNCH COMPLEX 39 IS 100% AVAILABLE FOR USE IN THIS MISSION
- REPLENISHING OF COMPLEX 39 FACILITIES (LOX, LH_2, RP-1, GN_2, WATER, ETC.) NOT CONSIDERED

FIGURE 7

Earth Launch Sequence and Schedule (Mars Flyby Mission)

RESULTS

PAYLOAD	DESIRED FIRING DATE	ACTUAL
S/C_9	9/4/75	4/28/75
S/C_8	9/5/75	6/24/75
TO_7	9/6/75	6/28/75
TO_6	9/7/75	7/6/75
TO_5	9/8/75	7/7/75
TO_4	9/9/75	9/3/75
TO_3	9/11/75	9/6/75
S-IIB$_2$	9/13/75	9/13/75
S-IIB$_1$ BACKUP LAUNCH	9/15/75	9/15/75

CONCLUSION

TANKER NO.	TIME IN ORBIT
TO_7	90 DAYS
TO_6	60 DAYS
TO_5	60 DAYS
TO_4	6 DAYS
TO_3	5 DAYS

COMPLEX 39 IS TIED UP FOR APPROXIMATELY 5 MONTHS

FIGURE 7a

Earth Launch Sequence and Schedule (Mars Flyby Mission)

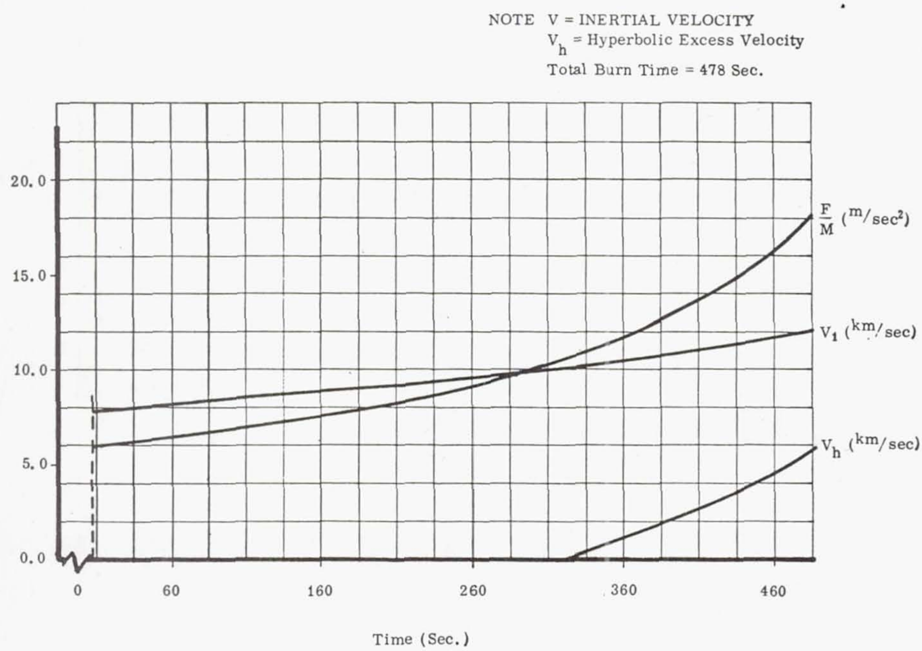


FIGURE 8

Trajectory Parameters for Earth Escape Thrusting 1975 Mars Flyby

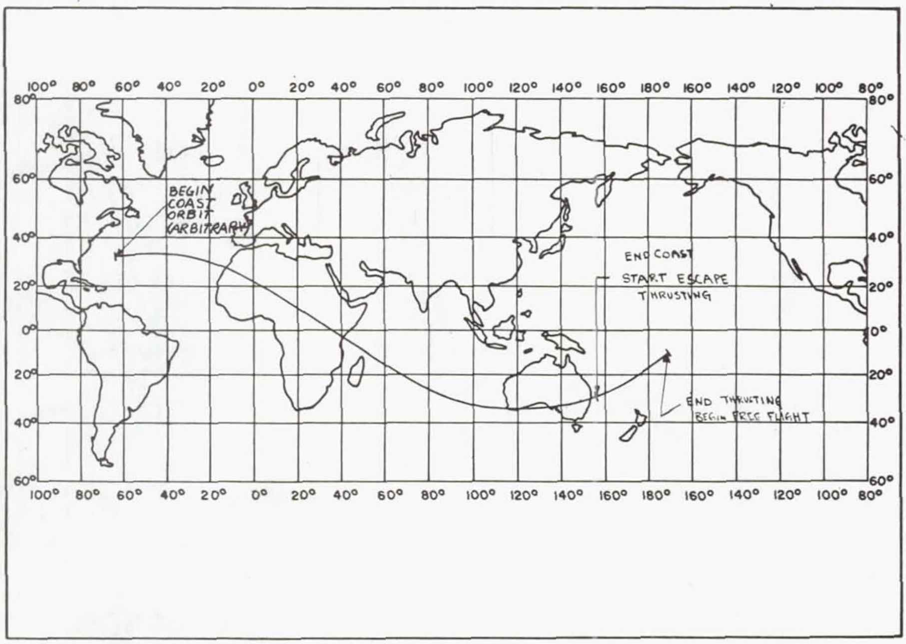


FIGURE 9

Coast Orbit & Escape Trace 1975 Mars Flyby

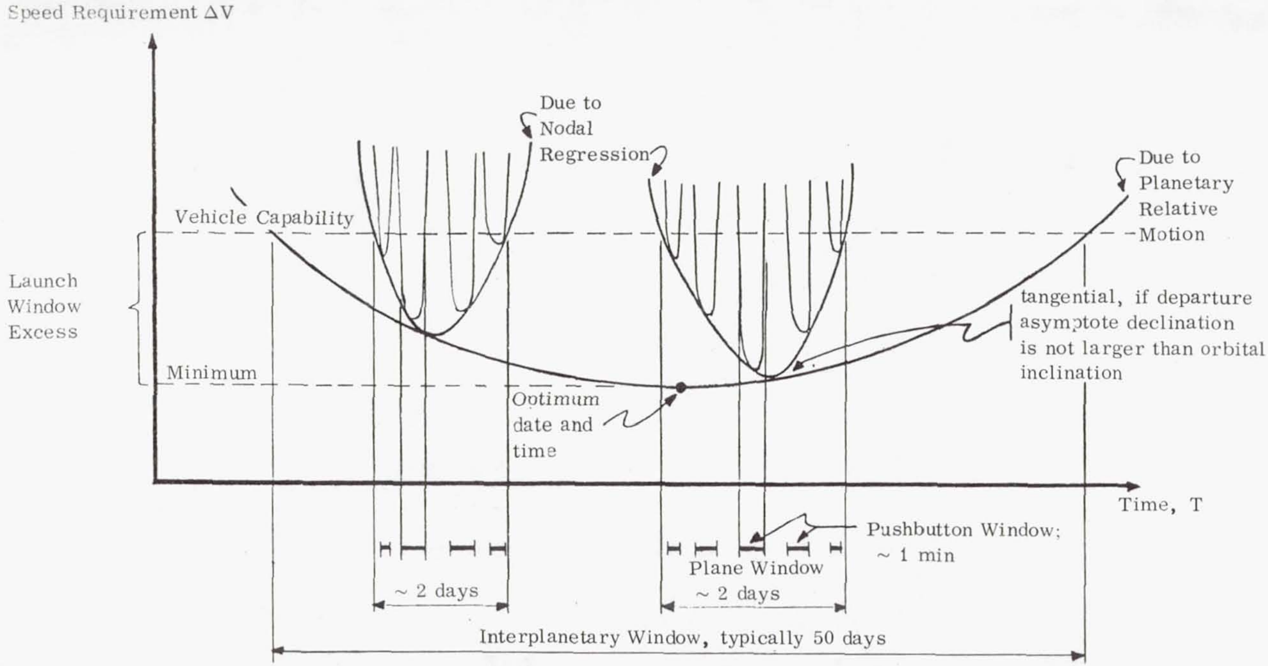
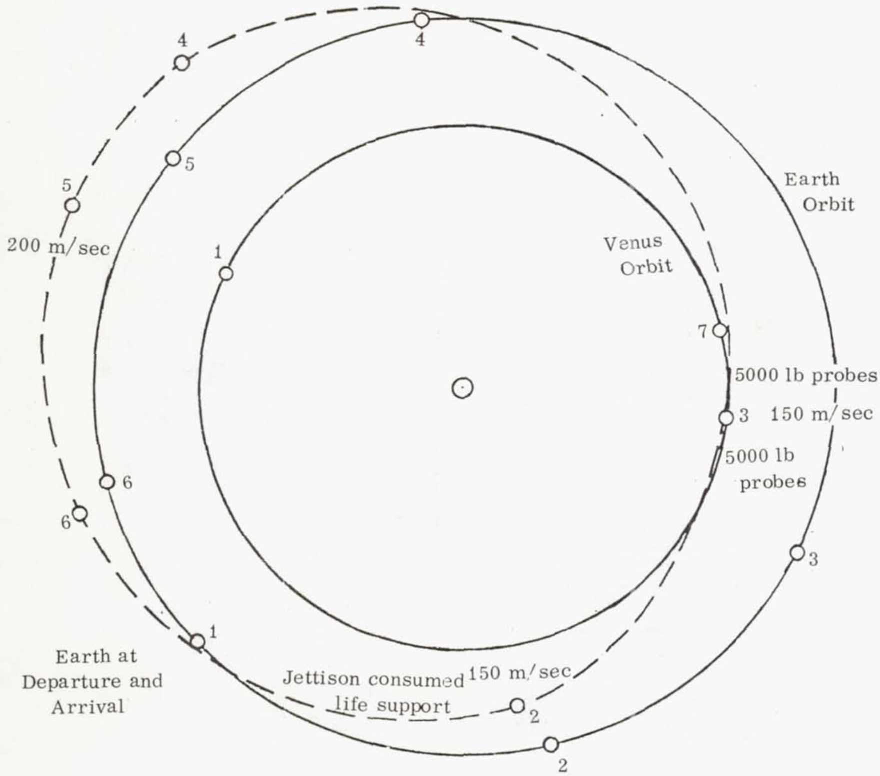


FIGURE 10

Interplanetary Launch Windows (Launch From Earth Satellite Orbit)



	Date	Communication Distance (A. U.)	ΔV km/sec
1	8-16-78	-	3.8±0.2 from orbit
2	10-15-78	.110	-
3	12-11-78	.395	-
4	4-15-79	.674	-
5	5-22-79	.324	-
6	7-24-79	.100	-
7	8-16-79	-	2.6±0.3 above parab.

FIGURE 11

Typical Venus Flyby Mission Profile

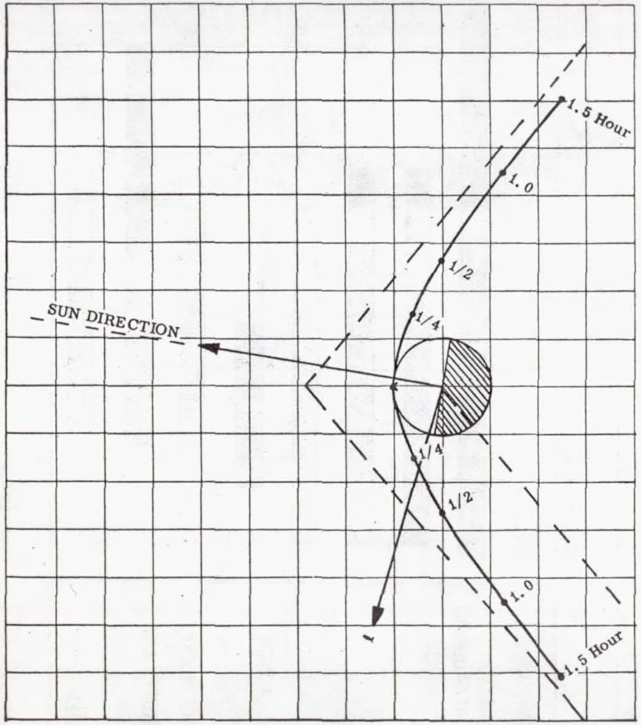
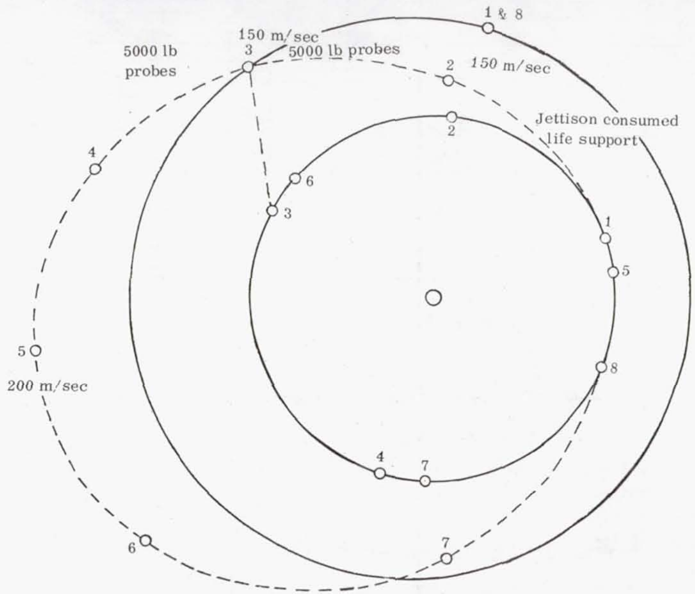


FIGURE 12

1978 Venus Fly-By Trajectory Hyperbolic Orbit Plane of Vehicle during Venus Passage Drawn To Scale: One-Half Inch Equal One Venusian Radius



	Date	Communication Distant (A.U.)	ΔV km/sec
1	9-26-75	-	5 ± 0.2 from orbit
2	11-30-75	0.24	-
3	2-3-76	0.86	-
4	5-16-76	2.35	-
5	9-13-76	3.21	-
6	1-21-77	2.07	-
7	5-31-77	0.353	-
8	7-25-77	-	4.1 ± 0.7 above parab.

FIGURE 13

Typical Martian Flyby Mission Profile

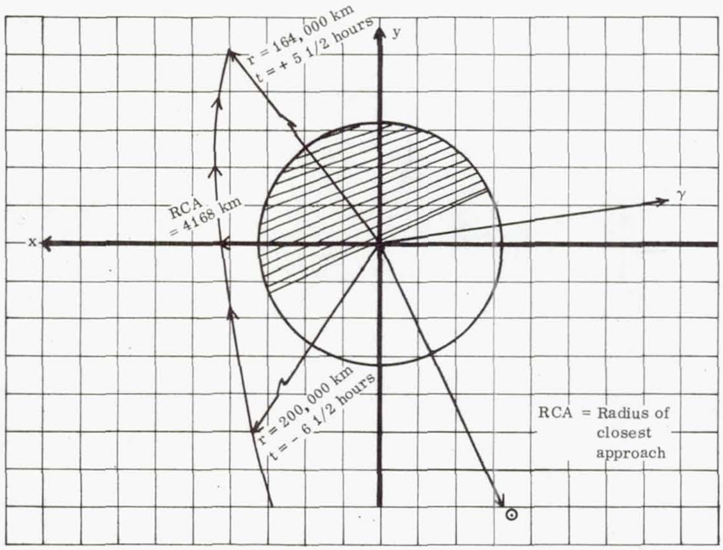


FIGURE 14

Vehicle Orbit Plane During Martian Passage 1975 Integrated Flyby

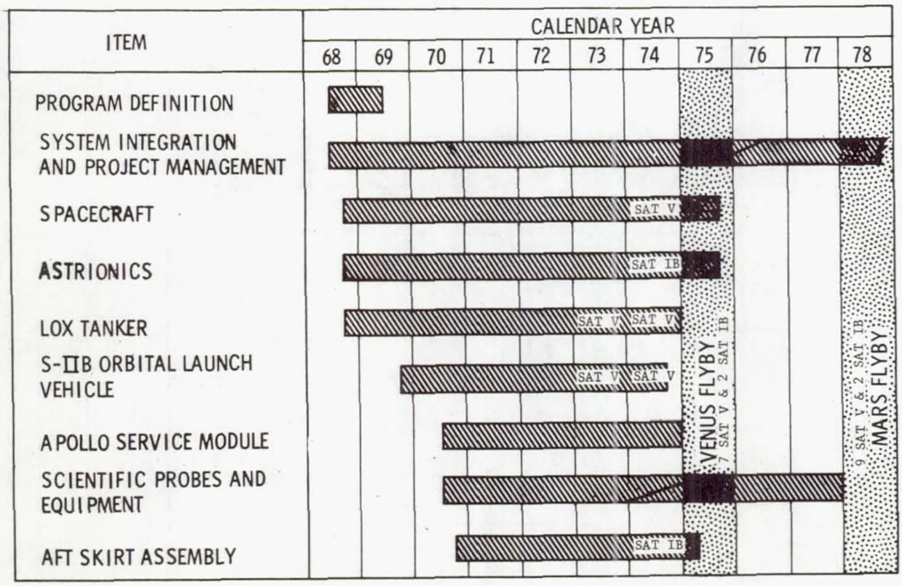


FIGURE 15

Design, Development and Test Schedule

Item	Number of Units	Average Unit Cost	Total Cost
Design and Development			
S-IIB OLV	-	-	420
Lox Tanker	-	-	380
Apollo Service Module	-	-	115
Astrionics	-	-	325
Aft. Skirt Assembly	-	-	165
Spacecraft	-	-	1,565
Scientific Probes and Equipment	-	-	440
Systems Integration and Project Management	-	-	340
Total	-	-	3,750
Venus 1975; Operational			
Saturn V	7	70	490
Saturn IB	2	22	44
S-IIB OLV	2	27	56
Lox Tanker	3	17	51
Aft. Skirt Assembly	2	15	30
Apollo CSM	4	72	288
Spacecraft	2	160	320
Total			1,279
Mars 1978; Operational			
Saturn V	9	65	585
Saturn IB	2	20	40
S-IIB OLV	2	27	56
Lox Tanker	5	17	85
Aft. Skirt Assembly	2	15	30
Apollo CSM	4	69	276
Spacecraft	2	160	320
Total			1,392

FIGURE 16

Design, Development, and Operational Costs

ITEM	FISCAL YEAR									
	69	70	71	72	73	74	75	76	77	78	TOTAL
DESIGN AND DEVELOPMENT											
S-IIB OLV		43	103	135	101	38					420
LOX TANKER	12	50	118	109	68	23					380
APOLLO SERVICE MODULE			17	40	40	18					115
ASTRIONICS	16	33	49	81	81	49	16				325
AFT SKIRT ASSEMBLY			16	33	58	41	17				165
SPACECRAFT	74	157	248	392	390	229	75				1565
SCIENTIFIC PROBES & EQUIP			24	71	57	77	86	57	53	15	440
SYSTEMS INT. & PROJECT MGT.	34	34	34	34	34	34	34	34	34	34	340
TOTAL DESIGN AND DEVELOPMENT	136	317	609	895	829	509	228	91	87	49	3750
OPERATIONAL											
SATURN V			54	157	128	183	220	152	141	40	1075
SATURN IB				7	21	14	8	19	13	2	84
S-IIB OLV				9	26	18	12	26	18	3	112
LOX TANKER				8	24	16	17	40	27	4	136
AFT SKIRT ASSEMBLY				5	10	10	10	10	10	5	60
APOLLO COMMAND & SER. MOD.					72	144	72	69	138	69	564
SPACECRAFT			32	64	112	112	96	112	80	32	640
TOTAL OPERATIONAL			86	250	393	497	435	428	427	155	2671
GRAND TOTAL	136	317	695	1145	1222	1006	663	519	514	204	6421

FIGURE 17

Obligational Funding Requirements (In Millions of Dollars)

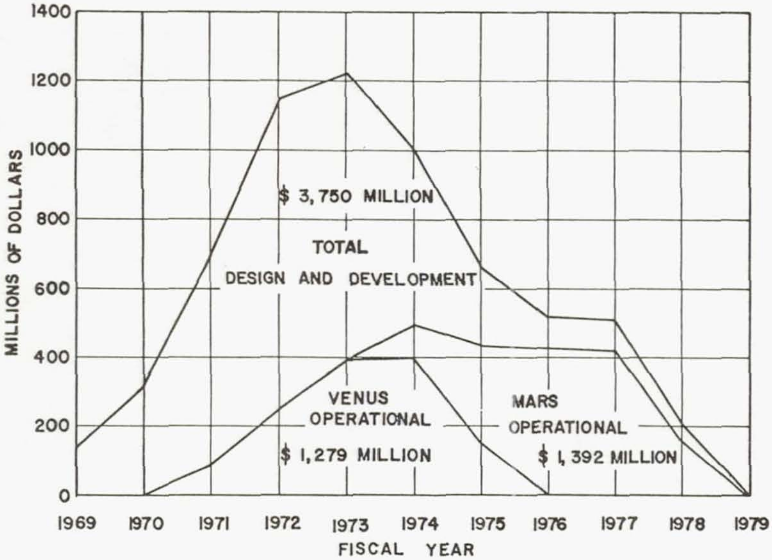


FIGURE 18

Annual Funding Requirement for Manned Venus and Mars Flyby Missions

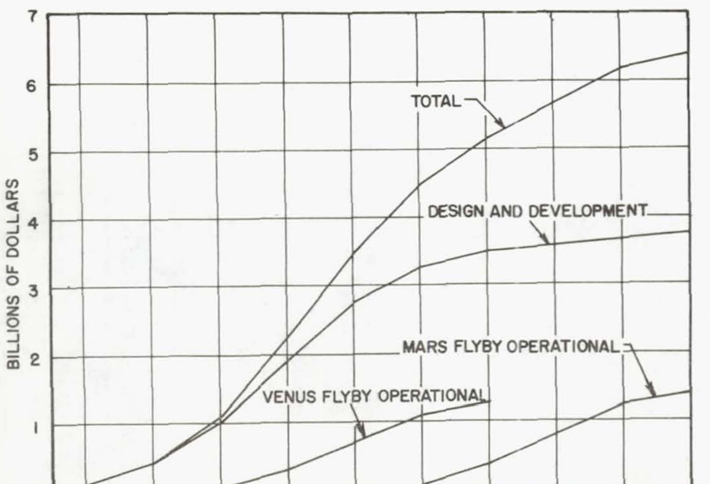


FIGURE 19

Cumulative Funding Requirement for Manned Venus and Mars Flyby Missions

	SPACE PROGRAM ALTERNATIVE			
	BASIC SPACE PROGRAM WITH VENUS AND MARS FLYBYS	REMOVE ALL MANNED PLANETARY MISSIONS	REMOVE MANNED MARS PLANETARY MISSIONS	REMOVE MANNED VENUS PLANETARY MISSIONS
PERCENT REDUCTION IN TOTAL PROGRAM WORTH	0	12	9	2
PERCENT REDUCTION IN PLANETARY PROGRAM WORTH	0	63	50	10

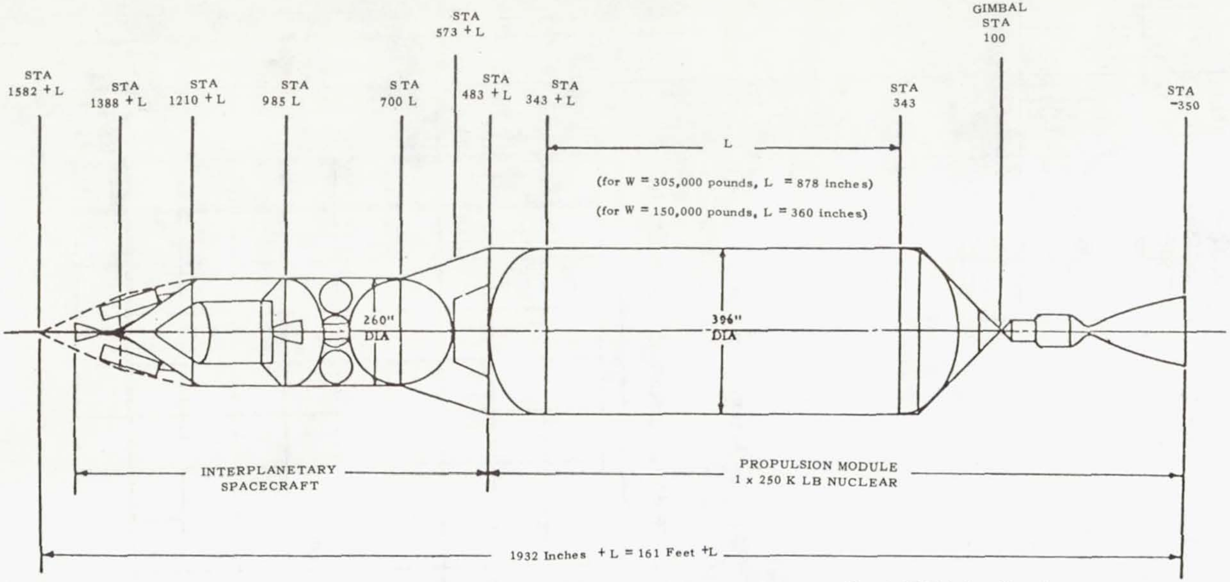
FIGURE 20

Relative Worth Reduction For Alternative Space Programs (In Percent)

		GENERAL				ORBITAL		LUNAR		PLANETARY				
OBJ. $[O_i]$	OBJ. WT. $[\theta_i]$	I_1	I_2	I_3		I_j					I_{60}	ΣP_{jp}	ΣP_j	
O_1	θ_1	$a_{11}q_{11}$											$\sum a_{ij}q_{ij} P_j$	$\sum_j a_{1j} \cdot q_{1j} = W_1$
O_2	θ_2													
O_3	θ_3													
O_i	θ_i					$a_{ij}q_{ij}$								$\sum_j a_{ij} q_{ij} = W_i$
O_{20}	θ_{20}													
		$\sum_i a_{i1}q_{i1}$											$\Sigma\Sigma aq$ iPj	$\sum_i \sum_j a_{ij} q_{ij}$

FIGURE 21

Typical Worth Calculation Matrix



An approximate relationship between length of nuclear stage and propellant (LH_2) loading is:
L (in) = $3.42\ W/10^3 - 153$, where W = propellant weight in pounds.

FIGURE 22

Nuclear Orbital Launch Configuration

N66 31505

PROBLEMS AND POSSIBILITIES OF EXPLORATION OF MARS SURFACE BY MANNED LANDINGS

By

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INTRODUCTION

The preceeding papers presented most of the facts relevant to the exploration of our two planetary neighbors, Mars and Venus. Mostly, they were concerned with unmanned excursions and observations. Now, we will discuss in what specific manner a manned exploration, more precisely, an exploration by men having physically landed on the surface of Mars differs from other means of investigations, and what special problems it poses. Doing so we will restrict our considerations to Mars for two reasons:

(1) It is safe to assume that landing on Mars will occur prior to landing on Venus. Hence landing on Venus will take advantage of the experience gained on Mars, and will also pose new problems different from those exposed here.

(2) We know so much more about Mars right now, and so little about Venus that it seems obvious that a landing on Mars is feasible for human beings, while we do not have the same good feeling at all about Venus. While we work towards landing on Mars, we also continue unmanned exploration of Venus, and we are sure that in the 80's, when we can hope to achieve our goal on Mars, many of those open questions for Venus will then be answered. Thus, we shall always be ahead in our knowledge of Mars in comparison to Venus, as it stands now. With the increased knowledge about Mars, which is needed for Venus, we shall finally be able to tackle this more remote problem. For unmanned missions, as fly-bys, this argument does not hold. To the contrary, Venus is more favorable than Mars in this respect, because of the more frequent close approaches, and the shorter flight time. Manned exploration does not necessarily require manned landing on the surface. Orbiting of a space craft staffed with humans in an orbit close to the planet has many obvious advantages over unmanned observations, and yields in many respects results as valuable as a physical landing of the observers. The information gained from manned orbiting to manned landing, in-so-far as observations are concerned, is much less than that gained from unmanned observations to manned orbitings or fly-bys. The main purpose, and advantage, of manned landing lie in the possibility of preparing for living at the surface, of colonizing the planet, etc. We understand that this goal is one of a more distant future, while for the near future, the collecting of pertinent data is the primary objective. For this reason, it is not necessary to step down to every spot on the surface in order to take advantage of Man's special capabilities. Even on Earth we do much thorough investigation without entering the regions physically. Sometimes, say on a battlefield, it is not possible to enter the area physically. And, further, let's remember that any exploration of Mars, say that done by Mariner IV on July 14, 1965, or any done by telescopes, or the naked eye in the entire past, is also a manned exploration, insofar as interpretation of the observations by the human mind is necessary and essential.

This is why we have to draw an artificial line and restrict our discussion to manned landings on the surface, not at the beginning at every spot on Mars, but at least at a single place. We do so because we feel that this very event, namely when a living, and thinking, human being for the first time, puts his hand and foot on the hard surface of Mars, separates two periods of investigation of this planet, and opens up a new, a more exciting, a more dangerous, and a more valuable era in this concern. Whenever this shall happen, in the 80's, as we suppose now, it shall not be man's first step in this direction. We assume, and we demand, that he has had experience walking through the craters of the Moon, and has probably turned the Moon's

surface into a kind of week-end resort place, at least for some daring Astro-hunters. He also shall have learned to live in outer space, far remote from the firm Earth, or even from a protecting space station. The landing on Mars will be just a new, however more exciting, event in a whole chain of adventures. We realize that we are, timewise, already somewhere up on this celestial ladder. Man has already left the Earth for a short while, and a few individulas have performed their first steps through the so-called "empty" space; but most of this adventure still lies ahead of us. At this place I would like to explain, and justify, the title of this presentation.

Firstly, it contains "possibility" of manned landings. The simple statement to whether such a landing is possible or not, is: "Yes, it is possible, right now, and even with our present available means." This is true in the same sense that a crossing of the Atlantic by a raft, or flying supersonic, or destroying a city by a bomb is possible. If we wish to put one of these possibilities into reality, we need hardware, and this in turn needs decisions, production, means of transportations, manpower, time, and money. But these needs do not restrict the possibility, they constitute a pure management-problem, which I will not consider here.

Secondly, however, any such realization has to face the consequences of the act, the dangers and the handicaps on one side, the advantages on the other side. These aspects are what are meant by the part of the title, "problems". We need to know those problems in order to overcome them, to avoid the dangers as much as possible, to protect the enterprise, and to draw full advantage of its result. The problem-areas, and the possibility are strongly interwoven. If some of the problems should turn out to be too severe for the decision-maker, the undertaking would then be simply undesirable, and is to be postponed at least. This would surely hold if we would start out with a crew for the surface of Mars this 27th day of August 1965. Our situation in this case would be exactly the same as that of one of the great discoverers, Columbus, or Marco Polo, or Nansen. Our present culture, and the incredible amount of effort involved, does not allow such adventure. We hope, nevertheless, that all those dangers and uncertainties will be sufficiently removed in the early 80's, say in 82, and we shall then be justified to venture the journey.

THE PROBLEM AREAS

1. The previous remarks state our first specific problem:

We have to think about manned trips, and landings, and we have to prepare them, far in advance, before we are able to rely on sound experience.

In previous times, it was the scientific custom to plan, and perform an experiment, to wait for its results, to interprete it thoroughly, to fit its result into an existing model, and then to design a new experiment on the firm grounds of the first one, and so on again, step by step. In other fields, say in politics, or in trade, Man was far more daring, and less patient. Many journeys to explore the sea, the polar region, far off countries, etc. have been performed simultaneously, without the necessary experience, and without sufficient safety.

In some respect, we are in the same position as those politicians, in spite of preparing scientific exploration. Time is pressing, whether we like it or not, whether it makes sense or not. Many non-scientific aspects and reasons, as for instance, prestige and military needs have to be considered. The preparations are very time-consuming, and costly. To be able to afford it at all, and achieve the main goals in proper time, we have to consider a host of differnet ways. We have to follow simultaneously many different roads, and ideas, and this all before the pertinent necessary knowledge is available.

It is clear enough that these circumstances bring along, not only new dangers and costs, but also the certainty of dead-end roads to be followed first, and to be abandoned after a while, of superfluous precautions and redundancies, and many other undesirable circumstances. But we can't help it.

As already said, other approaches to the over-all exploration of space are to be followed right now, and will continue in the future. We just learned a short time ago about the wonderful success of Mariner IV, and about the great surprises it has brought.

The most unexpected result was the large number of craters on Mars. I think, this is the first real surprise in Astronomy since the discovery of the second small planet, Pallas, in 1802 by Olbers, after Ceres had been discovered a year earlier. Although Olbers thought differently, none of the other astronomers dared to assume a second, even very small, body moving in the orbit of another one. Everything else up to now, at least in Astronomy, including the most spectacular technological achievement has been predicted, and expected. But nobody has predicted the existence

of craters on Mars in this abundance. One might presume that this new fact will be of great influence for theories on Mars to come and for the design of new landing devices.

It seems justified to expect in the future more surprises, and a lot of brake throughs, partially or wholly unpredicted, which will alter considerably our thinking about, and designing of landing vehicles. Our present concepts, in their wide variation, are just beginning ideas, based on an almost complete lack of experience.

If I said before that manned landing will open up a new period of exploration, this does not mean that it also will end the previous era of unmanned, say automatic observation. To the contrary, this latter one will have to continue, and to increase in amount and importance. Both means of investigation must run parallely, one implementing and directing the other one. This holds for all scientific fields, also on Earth, as for instance, Meteorology, or Biology, clearly show.

2. This remark indicates the second problem group, centered around the following questions:

a. What is the advantage of the physical presence of man on Mars, and in what respect is an automatic instrument as good, or better, than Man?

b. In what way can Man and Machine complement, and sustain each other?

Posing these questions implies also the following ones:

c. What are the dangers which Man faces going to Mars, and landing on Mars?

d. What way is less expensive in the sense of scientific return per effort? "Scientific" in the last question is used in its widest possible sense, and "Effort" includes time, money, psychological strain, etc.

Inseparably connected with these questions are the following critical ones:

e. Why do we send human beings to Mars; what do we want there, and what do we expect to find there?

f. In what respect does the traveling of men through space, and their stay on Mars, create peculiar problems and difficulties, and in what respect, on the other hand, will the presence of men facilitate the excursion?

I shall not be able to enumerate all the pertinent problems and aspects in all particulars, and, having stated some of them, I shall neither be able to answer them in a final way. The following selection must be a personal one for many reasons; the answers, as far as even an attempt can be made to answer them, can by no means be final. We will also keep in mind that most of the particular human factors to be discussed are valid, maybe for any other manned space exploration; only a few ones are specific for Mars alone and do not occur at other celestial bodies.

I shall not discuss the problem areas in the order listed above, but rather in a different grouping which I consider as being more methodical.

Let's first ask: Why do we strive for putting a man, or men, physically on Mars? The answer is simply, "We must."

Man is curious. This curiosity is called "research instinct" and might be a pure phychological problem. It might be, that we feel our power too limited if there exist places in the Universe where we cannot put our foot. We all feel so; we are never content merely to see pictures from Europe, say, on a traveling folder, or to listen to friends, telling about their advantures; we want, ourselves, to stroll through the Louvre, to stand at the Mermaid in Copenhagen, or to dine in Istanbul. In the same way, we just want to go to Mars, as well as to Pluto, or to the companion of a distant star. Only if our intelligence tells us clearly that we are not able to do it ourselves, or not yet, we would be, temporarily, content to listen to those who already ventrued the incredible. This is the least, we allow. Whether this drive is a curse, or a grace, is irrelevant. We are bound to it, in the course of our physical development from a pure chemical, low beginning towards an unknown height. If this generation would drop the idea, a following would pick it up again for sure.

In view of this "must", it sounds almost like an excuse to give reasons for space exploration. But here are some:

We are not able to understand our world fully, and not able to master it sufficiently, if we cannot have a close look at "other worlds", which means other possibilities, or particular solutions of this immense system of partial differential equations. Nobody knows his language, his country, his body, his desires, if he has no contact with other specimens of the same type, as foreign languages, the thoughts and behavior of other people and so on. Understanding the variability, the range, the extraordinary, and the improbable, alone yields a full understanding of our own environment. Only with the help of the Meteorology of Mars (or of another planet), shall we be able to comprehend the sequence of events inside the atmosphere of the Earth, to forecast the weather precisely, or to create a desired weather. with all

the practical implications to recreation, growing of crops, production of food, carrying out warfares, etc. As long as the ocean of air we live in is the only one we know, we live in the dark.

All the same holds true for Mineralogy, Geology, Geophysics, and all the various other fields of Science.

But as important and interesting as these fields seem to us at present, there still exists the very open question about the origin, and existence of life, especially of life on Mars.

Whatever we can learn about this burning problem will shed much light on the history, future, mechanics, and chemistry of this great mystery "life". This is the most exciting question of all, and Mars is more apt to furnish us with some answer, and more precise, and much sooner, than we can hope to receive them from any other source. Mars is still considered the most Earth-like celestial body in our Solar System, and perhaps the only one in existence at all.

Just to insert here a short remark:

The little, but nevertheless unique, information we got through Mariner IV has shown us to our surprise that even this most "homy" colleague of our human world is much less like our world than we believed. It seems to resemble much more our Moon, and it might well be that we shall be even more disappointed with coming results.

But disregarding this fact, especially since it is not well enough established, it might be appropriate to dwell a little on the subject of life on Mars.

Much speculation has arisen on "Life outside the Earth" throughout all the centuries. Philosophers, Theologicians, and many Scientists have proposed all types of answers to this problem, from:

"Life exists only on Earth, created here by divine decision, being unique in the Universe", to:

"Life is a general feature of the Universe, being omnipresent, and having generated as a random event everywhere where proper conditions prevail".

Everything between these two extremes has been stated and "proved". In order, for the second statement, to have a sound definition for "proper conditions", an extensive investigation of the regions of "ecospheres" around stars have been made, and probabilities of occurrance of life in different places of the Universe have been established. By the way, it has been found by such means that Mars is the only planet likely to bear life in our Solar System. (The fact that, in a neighborhood of 100 lightyears possibly 50 other Earth-like bodies might exist and shelter life, is of no interest for this lecture.)

If life is present on Mars (genuinely, or transferred there from Earth or from other places), it will be speculated whether it has gone through all the stages from most primitive to the intelligent, even to super-intelligent life. Strong voices have been heard for the real existence of superhumans; remember the controversies over the Martian Canals.

A moment of digression might be permitted.

History of Science tells us how Man has used his knowledge to force himself out from the firstly assumed central and unique position in the Universe, into a random-generated and random located chemical reaction, brother among many other brothers. Corpernicus finally pushed Earth out of the geometrical center of the world, and Astronomy has, in the course of time since then, proved that it is very likely that we are not alone in the Universe. Biology has joined Astronomy in its way, and also Philosophy teaches us the same.

But it might also be observed that exactly the same scientific achievements have resulted in showing that man does have a unique and lonesome position. If life exists somewhere else, and has developed from the same random events, in its span of development, however, none of the other "cultures" has as yet achieved what we are just doing right now, or have done in this century before. We could ask, "Why not"? "Are these other Supercreatures too uninterested in research"? Why then, are we the only ones blessed with our curse? Are the other Cultures already extinct? Why then, are we alone able to survive all those dangers we are exposed to, with a good expectation of surviving even much longer? Are we possibly so much older than other beings, or more capable, or faster in development? If our various "Earths" are equally old, and equally well fitted, why are there these extra features of our special brand of life? There are not many ways out of this trouble of the new loneliness.

However, as far as Mars is concerned, there has been found evidence for some life on Mars, independent of all probabilistic studies. Tichoff has observed spectral bands due to chlorophyll, and since then, some other spectral evidence of large, possibly living molecules has been established. Mariner IV has not yet corraborrated this findings. This probe was not designed for detecting traces of life, and such an observation was out of its range. With the same emphasis, we have to say Mariner IV has not disproven the existence of living material. Future probes, especially Voyagers will have the task to investigate this question. Later on, unmanned landings, and manned flybys will yield more precise data. We should be confident that before we land man on Mars we should have found out whether or not life exists.

But we can be sure only to a certain extent. We can detect for sure by preceeding probes some kind of life, for instance large signs of existing, or extinct life, traces of intelligent cultures, inactive, I mean immobile, life, like larger woods of lichens, bacteria in the soil etc.; all, if we are lucky enough, and if such types exist. But even existent life might escape detection by the usual means, as long as we do not land Man on Mars, and have him stay for a longer while, stimulating him to a very thorough and extensive investigation. Such life escaping detection could be for instance; bacteria deep in the soil, or restricted to single concentrated spots, very intelligent life hiding also deep in the ground, forms of life absolutely different from our aspect of life, created, say, this time, out of small, inconspicuous molecules, etc. I do not propose the idea here to believe that such a task, as exploring the unknown, especially without proper means, could be.

Searching for life on Mars or somewhere else also faces some other arguments, which are able to distort, and to bias the answer. The first argument is; "What is life after all"? Even on this small, and relatively homogenuous Earth, life has developed so many, different forms of shaping, of acting, of hiding, of adjusting, that the concept of "Life" has escaped a clear definition. This means in blunt words," define "Life" properly, and life exists for sure everywhere and at all times, even at the center of the Sun; define it differently, and life exists just on Broadway, New York. Saying "Life is what we find on Earth" from a virus to Albert Schweitzer, is a good and convenient definition most scientist will agree to. It dissolves the problem mentioned above, but still biases the answer. We might well have to change this definition.

The second argument is, as follows: Consider, a paratrooper jumping into the Viet Jungle. He will understand life as a Viet Cong Guerilla. Now, such a life might well be around our brave soldier, but he does not discover it. Life in all stages and forms is well adapted to conceal itself in the presence of a seemingly dangerous situation, and intelligent life is even more able to do so.

In short, detection of life is a very difficult and delicate subject. It can hardly be done with certainty by automats. To me, it seems, that a well trained, versatile, and patient man, provided with enough time, and facilities, is necessary to find an unquestionable answer. The importance of such an answer, whether it be "yes" or "no" is so obvious that it need not be discussed here. But it is also clear that a "yes" would be much more significant. It would need immediate and intelligent actions and decisions concerning further research, necessary protection, possible communication, means of not destroying, and means of destroying, and so on, which a machine never could do. A manned landing on Mars would be prohibited only in the case that previous observation would have revealed the existence of a kind of life, intelligent or just bacterial, which is dangerous to man, and which does not allow any kind of safe protection for the landing crew. From all we know now, it is very unlikely that "big" life does exist; bacterial dangers are very possible, but should be more serious only after a longer stay, and closer contact with soil, etc, and thus not for the first landings, where the crew could be sealed off from any external living influence, and would not depend on food growing locally. Despite this aspect, many other dangers for the crew might still prevail, but these should stem from the non-living conditions on Mars.

3. This little discussion of a vital problem has touched on the dual role man is playing in this Space Game.

Considering the obvious fact that space exploration is designed for man, with man participating, not only in an arm-chair way, but as one of the instruments proper, in other words, as a link in the chain of apparatus and events necessary and occuring during the journey, let's put this role in the following seemingly contradictory words:

Man is the purpose and means of the exploration; he is one of the many links of a long chain; he is the weakest link, and the strongest link; he is the fastest and the slowest of all the instruments, the most, and the least accurate one, the one which is able to provide most preotection, and which needs most protection. This all happens simultaneously.

The contradictions stated above are a reality. All problems of manned space trips are linked with one or the other of those diverse aspects.

Many of the facts to be mentioned are pertinent to any traveling through space, and hence are not peculiar to an early manned Mars landing. Again, many other problems will have been solved before the first landing is attempted.

We said, space traveling is an enterprise for man. This is true not only to serve man's curiosity, but for man's practical pruposes. There is, apart from scientific objectives, a wide variety of military and cultural reasons.

Just to mention one of them, the famous danger of the overpopulation of the world demands among other precautions preparing the possibility of colonizing Mars or other planets. This is surely not a plan for the very near future; it may be for the next century only, but preparation needs a lot of time, and should be begun soon. It will require a lot of traveling back and forth of all kinds of personnel, not only well trained scientist. Together with these people, another group of travelers will soon appear, the vacationists. In the 80's of this century, the cost of transport of a single man to Mars and back is estimated at about 10 million dollars, a tremendous amount. Only a very rich nation can afford to send a few men on such a trip, and they must therefore be well trained, and cared for. In, say, 2100, this cost might have fallen to $100, which would then make a reasonable selling price for a ticket of about $1000. With such a prospect, many people will be able to finance the journey, thus taking away another burden from the public.

Once technology has achieved this goal, traveling to and from Mars, landing on Mars, living there for a limited time etc. must be possible for every person with an average health and physical ability. Up to now, only 1% of 1% of 1% of 1% of the population of this country would be allowed to undertake this adventure; they are comparable to the courageous pioneers of the 18th century. But this nation would not be this nation, if it would always have needed a daring pioneer ready to shoot and to hide, to cross over from New York to Los Angeles. Instead, every space trip has to become, in time, easy, routine, and safe for everyone.

A host of problems pertain to this tremendous task. Many of them concern the traveling proper, lift off from Earth, high acceleration, state of weightlessness, confinement to narrow living quarters, lack of minor facilities, loneliness, homesickness, companionship to a few persons for a long time, fear of expected or unexpected dangers, rediation, soft landing, entry through an atmosphere, new lift off at return, and many others.

Space medicine has taken care of those problems, and is well on the way to solve and remove, quite a few of them. Other, yet unknown ones, might arise later. When the first manned landing on Mars will be attempted, trained astronauts will be available who have experienced, and have overcome, almost all of those dangers and strain.

Only a few facts are specific to the first manned landings and walking over the surface. Fortunately one can hope that these problems are minor ones in comparison to those mentioned above.

The reason is, again, that Mars is more Earth-like than the Moon in many respects. Mars seems to have no radiation belt, and a very low magnetism; shielding against radiation is therefore only necessary for the region close to Earth and on the trip, and not at the surface itself. This makes everything a lot easier. It is true of course that the lack of a belt, in connection with the thin atmosphere exposed the surface to a much greater amount of unfavorable UV, cosmic, and plasma radiation from the Sun and from outer space, and makes it also subject to strong and unexpected variations with time. For the first landers therefore, an elaborate space suit might be needed. But this is also true of the Moon landing, which should already be conquered at that time.

Traversing the atmosphere of Mars is less dangerous than traversing the atmosphere of Earth, because of the smaller density. On the other hand, parachutes, or gliding vehicles wouldn't work like on Earth. This poses a lot of technical problems in design of landing vehicles etc, but none of them are really tough especially after exact data concerning the air mantle of Mars is known from orbiters and hard landers.

It is not possible for a human being to breathe the air of Mars directly because of the seemingly total lack of oxygen. One must use devices like those used on the Moon, where the roving crew experiences a total lack of gas. Again, this is no real problem, only inconvenient. But the air could to the contrary be dangerous. The presence of CH_4, with danger of explosions, is not disproved. The air could be in heavy motion, thus handcapping or even endangering human activities outside of the sheltering laboratory. The air might be loaded with sand of small grain, penetrating through suits and instruments. We have no real difficulty in protecting the crew against these hazards, but we have at least to be aware of them being costly and unpleasant, and they hamper quick activities, disturb the sight from a landing spot etc. Proper design of protection again needs collection of enough and precise data prior to landing. We understand also that automated instruments would suffer some of these handicaps also, but would be uneffected by other ones.

Having landed safely on the surface, and being protected in the above sense, the explorer finds an environment not too different from that one at certain places on Earth. It is true that it will be unpleasant there for the first time, but man has adjusted on Earth sometimes to more severe conditions. At noon-time in midsummer he will observe a daylight roughly as dim as on a cloudy November day in our country. There will be a dark, violet sky, with some stars in it, and a sharp, relatively brilliant sun. High mountain places, or the polar regions on Earth show about the same features. The soil is supposed to be a loose, yellow sand, like in the midst of our deserts. Very likely, some places look quite different, they may be swamp like, or covered by moss like plants, or covered by snow, or, we might even find smaller ponds of open water. We thought a while ago that the surface should be relatively smooth, basically plain, with soft slopes only. That part of the surface, however, which was covered by the TV camera of Mariner IV shows an abundance of rocky features, craters, high peaks (13,000 feet, but of course not as rugged as the Moon), and slopes up to 10 degrees. It is not justified, of course to generalize; we need more and better information. All this could happen on Earth too. Having a cozy Mars house on the ground, our pioneers could easily adjust, and enjoy their stay. For how long, we don't know yet. Even the best trained, and most willing persons, even in most pleasant surroundings and conditions, become psychotic to a more or lesser extent after some time. It is to be expected also that some time after landing the effects on body and mind of the heavy strains piled up during the trip, and driven back by the excitement of the first experiences after landing, will break out in this more quiet condition, and create very dangerous situations. The crew, and the station must be well prepared for all these possibilities, also for the loss of operating powers of persons affected badly. At later stages of space travel, medical aid stations, repair shops, rescue teams, recreation facilities, etc. will be available. But for the first trip, the world has to be prepared to accept some total losses of men and material, since we cannot be prepared for every possible event.

4. There is another far reaching difference between unmanned, and manned exploration of Mars, apart from the aspect of protection.

Man, is as already said, one of the instruments used both for performing the trip and executing the exploration. He is the most versatile, the fastest, the most accurate, and also the most restricted, the slowest, the coarsest, of these instruments.

No automat is as versatile as man. He is a thermometer, a telescope, a hygrometer, a tape recorder, a computer. He is programmed for the expected as well as for the unexpected. This fact makes him indispensable for space exploration. It also gives him the unique power of making decisions right on the spot, of changing the whole program, of facing, and mastering the unexpected. His decision might prove wrong, and even might cost him his life; but it is a decision anyway. The automat confronted with the same situation just breaks down, and waits for repair. It is clear enough that as long as the unknown threatens the early expeditions need such decision makers. Once this unknown is restricted in range and effect, automated robots do a much better job than man. Automated weather stations yield better and faster results than man, without getting tired, worn out, and bored, once they know what to measure. But for centuries past, and a long time to come, personal observations of meteorological parameters were, and will be, needed to get out all the advantages of impersonal instruments.

It is a consequence of these circumstances, but also inherent to the structure of man, that he is also the most inexpensive instrument in existence, and this includes money, weight, and volume.

To show this again in an obvious example: a man of 160 pounds, only two pounds of them thinking, can drive a truck safely from New York to Los Angeles. I doubt whether a million megaton robot could do the same, avoiding all the traffic traps, old ladies, young children, cops, improper road signs etc. And please, compare the costs!

Man will land on the surface, look around, find the most interesting path to proceed, smell an unexpected gas, hide before a yet distant sand storm arrives, detect a clever mimikry, or a slow and hesitating motion. He will then dicide to stay longer or to come back, to flee before an enemy destroys his memory, defend himself against many dangers, and work for quite a while without immediate supply of energy. The machine just stands there and does its job, very precisely indeed, but without fear and hope.

The observations made by man are precise enough to assure his survival, and to direct the instruments how to furnish more accurate data for a better theory. Man is fast enough to master many situations; whenever he is too slow, like in numerical computation, the work can be predone or postponed. Man has an extremely wealthy memory, and a very easy way to improve it with a little piece of paper.

Defending man's role in such a way is justified, but will inadvertedly bring out his weaknesses too. Some were mentioned previously in his need for protection. His range of living, and acting, is so narrow. He cannot stand high or low temperature which would not even affect the instruments. He cannot see in the dark, while a small vidicon can. He feels bad if his environment, say air, is not what he is accustomed to. His memory fails often, or even deceives him. He is so easily distrubed by a single word, the accustical energy of which is negligible. And, above all, he is so easily bored, and always hostile to his fellow observers, at least in some remote chamber of his heart.

In comparison, instruments work under all conditions. They do not mind the presence or absence of a fellow thermometer, and need not be entertained at regular intervals.

The picture I have just painted for you shows light, brilliant spots covered with dark, greasy oil. Don't be afraid. We know the solution to make everything clear and pleasant. It is <u>cooperation</u> between man and machine. Put the man in the right place when he is superior to the machine, and let the machine do that job for which man is not fit. Man and machine have to cooperate on the surface of Mars as they have to on Earth throughout all times. They have to alternate and implement each other in all needs for preparing the trip, getting the excursion done, and later in evaluating the results. Man has to do a lot of thinking to create the proper machine, and to place it in the right condition, in right time, on the right spot. The machine does its job, while the man supervises it, changes its course, if necessary, and the machine goes on with the new task without asking nasty questions. Our man driving from New York to Los Angeles is also involved in a close cooperation with his automobile and all the mechanical devices pertinent to the trip, like traffic lights, machine guns, bicycles.

In this sense it will be fully appreciated that the first explorations of Mars have to be automated ones, and that later, but at quite early steps, Man comes in, and that he himself will be replaced a while later again by untirable, and unfailing machines, in a similar way as the pilot of a plane, or the captain of a battleship in action might be replaced by automats.

5. Let us not forget another cooperation, as important as that one between man and machine, the international cooperation.

What sense would there be in all the efforts of careful planning, preparing, and designing a trip to Mars, in training all those valuable astronauts, in tirelessly collecting small pieces of expensive information, if some cosmonauts do exactly the same, and faster. If we would decide to wait, say for 50 years, and use their results to make then a routine pleasure trip right from the beginning, our gain in knowledge would be exactly the same, with no money, lives, or effort spent. Of course, they could think the same way, and wait for us. If both choose this approach, nothing at all will happen. If both try independently, there is on both sides a tremendous waste of fortune, including human brain power. If one should stay home, and the other proceed, which one is it that is to retreat?

Cooperation is the only true solution. It could bring us forward in a fraction of the time, with a fraction of the money to be spent, and bring us much further out into space than anyone can achieve alone. Only a true cooperation of the entire world with all its facilities and resources can promise real success, without too great a burden for the single taxpayer.

There is little hope at the moment for such an agreement, but man might nevertheless advance also in this respect in the near future, and prove that he is more clever than it seems now.

6. A close cooperation and a thorough surveillance of the other's activities is especially needed in one area, which has not been mentioned yet.

Out first concern is to detect whether there is life on Mars or not, and if yes, what kind of life. This demands categorically that we do not bring life there ourselves, except that of the crew of course, which we want to have back unharmed anyway. But it would be too easy to plant some life on the planet's surface, since viruses, bacteria, insects etc. creep so easily into everything. We would have no difficulty in detecting a blind passenger in the space ship, or a dog in one of the closets. But decontaminating a space ship completely, and with positive safety, is something quite different, especially since some kind of bacteria are vitally necessary for human life, or for food for humans. Much thought has been given to this problem, and we might be sure that, to all conceivable extent, everything is sterile which reaches the surface of Mars the first time.

However, there are a few points to worry about. Firstly, what would happen if life would generate on or inside the space ship after it has left the launching pad? This is unlikely but not impossible, since we consider life as a product of randomness, and since we can hardly draw a safe line between living molecules and "true" life. Under the conditions in space or on Mars, different from those on Earth, such a mutation is not at all impossible. But it would, in this case, be very hard to discover this event.

Secondly, we do want to learn how different types of plants, animals or bacteria etc. thrive, or perish in the new environment, similar to how we have sent up apes, dogs, and other animals, in order to gain some experience valuable for ourselves. We can dissolve this controversy by asking that the first trips should avoid contamination by all means, while at a later stage we might conduct different biological experiments.

Thirdly, and more important, much more dangerous than contaminating the Martian surface with earthly living beings is to contaminate the Earth and its population with species developed on Mars. In the case of Martian virusses (and Mars might have allowed the creation of entirely different enemies to Man) we would have hardly any opportunity to detect them, and almost surely no possibility of prohibiting them from penetrating into the space ship, or into the bodies of the crew. A quarantine before returning to Earth would help a little, not completely. The nature of the beings would not be known, and there would be no means of destroying them. Even after a lapse of a very long time, the danger would not be over. We have had many experiences in the past, where such virusses have been found to be resistant against all unfavorable conditions, and time. Above all, we know how dangerous they are to the human race.

This is a very difficult fight against a completely unknown (maybe non-existant) enemy. We can only hope, that such a danger will not occur. Our past experience leaves, however, very little space for such a hope.

Fourthly, and finally, what if the other party in the game should contaminate Mars before we reach and secure it? This could be done by mistake, by mischief, for military purposes, and for many other reasons. It is not necessary that the other parties arrive earlier than we. It would suffice for them to crash biological bombs there, which could be done tomorrow. Contamination could be achieved by vires as well as by soldiers.

We understand that more than in other respects, cooperation and surveillance is vital.

This concludes this short presentation. Its purpose was, briefly, to point out a few of the problems we have to face, without going into technical details and to show that we are well on the way to solving them, and we are well prepared to face everything in order to achieve our great goal: conquest of space.